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Flight Report Vehicle 317/2020/3020
Delta Program-Mission No. 7
Tiros D Payload

IV

MARCH 1962

DOUGLAS REPORT SM-41505

MISSILE & SPACE SYSTEMS DIVISION
DOUGLAS AIRCRAFT COMPANY, INC.
SANTA MONICA/CALIFORNIA

DOUGLAS

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ABSTRACT

This report presents a summary of the objectives, results, and analyses of the countdown and flight of the Delta Vehicle used to launch the Tiros D Satellite.

The primary objective of the flight, which was to inject the Tiros D Satellite into a specified orbit, was achieved.

This mission was the seventh in a series of twelve space research vehicles to be launched under the original Delta contract.

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SUMMARY OF REPORT

Evaluation results are presented in this document for the launch operation and the performance of the Delta vehicle up to payload injection.

The primary mission objective was to inject the Tiros D spacecraft into a prescribed orbit. This objective was achieved.

Delta vehicle S/N 317/2020/3020 was launched from Pad 17A, AFMTC on February 8, 1962. This was the seventh mission of the Delta Program and the sixth consecutive success.

The launch was accomplished on the third attempt at the conclusion of a normal countdown.

All systems performed satisfactorily and all functions of the spacecraft are reported to be operating normally.

1. INTRODUCTION

The primary objective of the Tiros D experiment was to obtain meteorological data from a satellite circling the earth in a nominal 380-nautical mile circular orbit. All mission objectives were achieved.

The satellite was provided by the National Aeronautics and Space Administration (NASA). It consisted of an 18-sided polygon, with the top and side surfaces covered by solar cells. The satellite was designed to view the earth's cloud cover by means of one wide-angle camera and one intermediate-angle camera. It also contained infra-red experiments similar to those contained in Tiros II and III.

All flight times noted in the text are seconds after liftoff.

Preliminary Flight Test Report No. 7 (Reference 1) was used in preparation of this report.

2. MISSION AND VEHICLE OBJECTIVES AND SUMMARY OF THEIR ACCOMPLISHMENT

The following objectives were taken from reference 2.

2.1 Primary Objective

Demonstrate satisfactory payload injection into specified orbit.

The payload was injected into an orbit having an apogee of 462 n. mi., a perigee of 389 n. mi., an inclination angle of 48.298 degrees, and a period of 100.397 minutes. These parameters were within the tolerance of the mission and the objective was considered to have been achieved.

2.2 Secondary Objectives

Demonstrate satisfactory first stage performance.

Demonstrate satisfactory second stage performance.

Demonstrate satisfactory third stage performance.

Demonstrate satisfactory operation of all checkout and launch equipment.

All objectives were successfully accomplished.

3. DISCUSSION OF RESULTS

3.1 Summary of Flight Events

3.1.1 Flight Mechanics

This section describes the spatial flight path of the Delta Tiros D vehicle and discuss any differences from the nominal which may have affected the achievement of the primary and secondary test objectives.

The nominal used as basis of this analysis appears in the DAC Detailed Test Objectives (reference 2). Although the DTO incorporated nominal third stage engine performance and weight values, evaluation of actual performance was based on the actual third stage unit (SV-117) tag engine performance and weight values. Nominal and actual sequence of events which occurred throughout flight are compared in Table 1. A comparison of vehicle performance parameters and trajectory characteristics of the nominal and actual flights are shown in Table 2.

First Stage

Performance of the first stage was satisfactory. The actual trajectory compared to the nominal in the vertical plane (illustrated in figure 1) was lofted and further downrange in time. A lower than nominal liftoff weight and a higher than nominal thrust were the predominant factors responsible for the differences in the trajectories.

The horizontal projection of the nominal and actual trajectories are displayed in figure 2. Tracking data show the vehicle to the right of nominal soon after the first stage roll program had been completed, and continued to diverge from nominal to approximately 90 seconds. Reconstruction techniques revealed that winds had little effect on the cross-range deviation between the two trajectories during the early portion of flight. This is a good indication that a sub-nominal roll program was achieved, since yaw gyro drift and yaw thrust misalignment were small. Winds tended to have more effect on the trajectory during the high wind region (60-80 seconds from liftoff). Maximum wind velocity was 187 feet per second at 40,000 feet, and the maximum wind shear during the high wind velocity region was 16 feet per second per 1,000 feet at 46,000 feet. Wind direction was predominately from the west. BTL guidance system, initiated at 88.9 seconds, corrected the flight azimuth and the remainder of first stage powered flight was flown on a path parallel to nominal.

Figure 3 shows the actual and nominal flight path histories (measured with respect to the relative velocity vector). The actual flight path was lower than nominal during the first 80 seconds of flight. Winds and an increasing dynamic pressure seemed to depress the flight path angle to a maximum deviation of 1.5 degrees at 70 seconds after liftoff. As the winds subsided and dynamic pressure began decreasing, the actual flight path angle approached nominal. BTL guidance system effectiveness was demonstrated as the flight path angle was corrected, and the remainder of first stage flight was flown on a near nominal flight path.

The actual relative velocity (illustrated in figure 4) was higher than nominal throughout first stage flight. This was, again, attributed to a higher thrust and lower liftoff weight than nominal. Instantaneous impact predictions for the nominal and actual trajectories are shown in figure 5. The higher and faster than predicted trajectory resulted in first stage impact approximately 100 miles downrange from nominal. The crossrange deviation of the first stage impact point from nominal was small. Main engine cutoff was initiated by fuel depletion at 159.3 seconds, 0.3 seconds earlier than nominal. Propellant utilization was 99.95 percent, compared to 99.6 percent for nominal, and the average weight flow increased 0.38 percent over nominal. Reconstruction methods, indicated overall booster thrust to be 1.12 percent higher than expected, and the specific impulse was 0.77 percent greater.

Normal vehicle disturbances occurred during the high dynamic pressure region as indicated by telemetry data. Preliminary post flight analysis indicated that a booster fin may have come off during the flight through the Mach 1 region, but a more extensive analysis indicated that there was insufficient evidence to corroborate the notion of a fin loss. Disturbances in telemetry data are readily seen at about 48 seconds after liftoff (Mach 1 region) particularly in roll attitude and yaw attitude error channels, but only slightly in the pitch attitude error channel. These disturbances were greater than those experienced on previous Delta flights, and might substantiate a fin loss. However, accurate timing of the flight films sequencing clearly showed that no objects were jettisoned from the vehicle during the flight through this region. However, the films did indicate that an object was shed at 76 seconds, but it is believed the loss was not a booster fin since there were no disturbances shown on telemetry at this time and the vehicle was beyond the maximum dynamic pressure region where the greatest aerodynamic loads are experienced.

During first stage burning the maximum pitch rate, pitch attitude error, and main engine pitch position were -1.05 degrees per second at 91.5 seconds, 1.35 degrees at 75 seconds, and 1.13 degrees at 64 seconds, respectively. Maximum yaw rate, yaw attitude error, and main engine yaw position were 1.6 degrees per second at 50 seconds, 1.12 degrees at 62 seconds, and -1.15 degrees at 61.5 seconds, respectively. Pitch, yaw, and roll rates at separation were 0.25 degrees per second, -0.20 degrees per second, and 0.40 degrees per second, respectively. There were no noticeable adverse effects on the trajectory resulting from these disturbances.

Second Stage

Second stage ignition was commanded by the second stage programmer at 163.7 seconds followed by BTL guidance cutoff 108.4 seconds later. Steady state thrust and specific impulse as determined from tracking and telemetry data were 7500 pounds and 265 seconds respectively. These values indicate second stage performance was slightly less than predicted inasmuch as nominal thrust and specific impulse were 7551 and 265.3 seconds. Propellant remaining at second stage engine cutoff (SECO) was about 6.0 per cent of the total propellant available for impulse. A summary of second stage performance parameters compared with the nominal values is shown in Table III.

The actual and nominal trajectories in the pitch and yaw planes are illustrated in figures 7 and 8. For constant timing marks, an actual trajectory is shown to be lofted and ahead of the nominal trajectory throughout the second stage burning phase. Altitude and surface range deviations at SECO were 5.7 and 6.2 nautical miles respectively. These deviations are due in part to the better than nominal performance of the boost vehicle during the first stage burning phase. Maximum deviation of the actual trajectory in the yaw plane was approximately one nautical mile to the right of nominal. At SECO, the actual and nominal trajectories were nearly coincident in the yaw plane.

The actual and nominal velocity and flight path angle histories (with respect to the earth relative velocity vector) are shown in figures 8 and 9 for the second stage burning phase. The actual velocity curve is higher than nominal illustrating the 408 feet per second excess velocity at MECO. The velocity deviation decreased during second stage burning and at SECO was about 45 feet per second greater than nominal. The velocity deviation is attributed to the combined effects of the excess velocity at MECO and the lower than nominal performance and lofting of the vehicle during second stage burning. The velocity deviation does not reflect the accuracy of the BTL guidance system since the SECO velocity required for a successful mission depends in part, on spatial position of the vehicle.

The actual flight path angle was somewhat higher than nominal throughout second stage powered flight. At SECO the deviation was about 0.5 degrees above nominal. Second stage cutoff was initiated by BTL guidance command at approximately 272.1 seconds after liftoff. BTL open loop commands previous to SECO were 1.278 degrees left and .325 degrees up in yaw and pitch, respectively.

Telemetry data indicated that the second stage engine chamber pressure dropped by 90% within 0.6 seconds after the command was given for SECO. Assuming a linear thrust decay during this time interval results in a cutoff impulse of 2550 pound-seconds. The average cutoff impulse obtained from the last six Delta firings is 2510 $\pm$ 590 pound seconds which is in reasonable agreement with the expected vacuum impulse of 2400 $\pm$ 610 pound-seconds.

Attitude deviations in pitch, yaw and roll between MECO and first-second stage separation were small in magnitude and the attitude rates at ignition were essentially zero. Attitude deviations at ignition were less than 0.7 degrees and were corrected by the second stage control system within 8 seconds after separation. Except for a few small oscillations at initiation of BTL open loop steering, pitch and yaw deviations were negligible throughout the remainder of second stage burning. A roll attitude error of about one degree occurred at the time of nose fairing jettison (184 seconds from liftoff). This disturbance was not evident in pitch or yaw.

Telemetry data indicated that a constant counter clockwise disturbing torque was present during the second stage burning phase. This disturbance resulted in one sided control jet operation. Previous Delta vehicles exhibited similar disturbing torques, the direction from vehicle to vehicle being random. The roll torque is believed to be the result of swirling engine exhaust gases.

The actual and nominal instantaneous impact predictions (IIP) for the second stage are shown in figure 5. The predicted impact for the second stage was approximately 88 miles downrange of the nominal illustrating the effect of the greater than nominal altitude and range at SECO. Crossrange deviation of the impact point is approximately 12 nautical miles to the right or south of the nominal IIP trace. The downrange and crossrange deviations were well within the three-sigma guidance inoperative ellipse specified for this flight.

The destruct line constructed on the map, figure 5, illustrates the location on the IIP trace, and the time at which a premature SECO would have resulted in a third stage land impact. Had SECO been commanded at approximately 256 seconds or less, compared to 262 for nominal performance vehicles, a third stage land impact would have occurred and an orbit would not have been achieved.

Coast Phase

The coast time of the second stage vehicle was 434.7 seconds or 0.1 seconds longer than nominal. This difference is the result of the combined effects of the slightly shorter than nominal second stage burn time and slightly shorter than nominal time interval from second stage programmer start (MECO) to second-third stage separation. The deviation in coast time from nominal was small and therefore had little effect on the payload orbit.

The jet control operation during the coast phase was normal as indicated by telemetry. The third stage spin rockets were fired at 705.0 seconds resulting in a spin rate of 116 RPM. The expected spin rate was 126 RPM ± 10 percent.

Pitch and yaw attitude rates immediately following separation were small (less than 0.1 degree per second) indicating a smooth second-third stage separation.

Third Stage

The third stage ignition time delay relay was initiated by the second stage programmer 2.0 seconds before second-third stage separation, and the third stage engine ignition occurred 720.5 seconds after liftoff.

The performance of the third stage engine was higher than nominal, as determined from an analysis utilizing tracking data and payload orbital data. The total impulse was 116,650 pound-seconds which compares to a nominal value of 116,060 ± 618 pound-seconds. A summary of third stage performance parameters is shown in table II.

The Tiros D payload had the following injection conditions, i.e., conditions at third stage burnout.

Inertial velocity	24,652 feet per second
Inertial elevation flight path angle	-0.375 degrees
Inertial azimuth flight path angle	66.54 degrees
Altitude	399.4 nautical miles
West longitude	53.38 degrees
North latitude	43.69 degrees
*Vehicle centerline elevation angle	3.04 degrees
**Vehicle centerline azimuth angle	64.49 degrees

* θ_L', Angle between the vehicle centerline and a plane perpendicular to the radius vector from the center of the earth to the vehicle, positive for vehicle pointing away from the earth.

** ψ_L', Angle between the local meridan and the projection of the vehicle centerline onto a plane perpendicular to the radius vector from the center of the earth to the vehicle, positive clockwise from true north.

A comparison of the achieved orbit parameters with the BTL pre-flight nominal and the BTL preliminary post-flight orbit parameters is shown in the following table: (Altitudes are based on an earth radius of 3438.5 nautical miles.)

PARAMETER	BTL PRE-FLIGHT	BTL ACTUAL SECO AND NOMINAL THIRD STAGE	ACHIEVED
Apogee (n.mi)	380.0	403.53	461.65
Perigee (n.mi)	379.9	394.40	389.18
Inclination (deg)	48.3	48.32	48.30
Eccentricity	.00001	.00119	.00938
Period (min.)	98.64	99.38	100.40

* Computed from minitrack observations at the NASA Computing Center - Epoch 8 February 1962, 1257 UT.

The larger than nominal orbit computed by BTL using actual SECO conditions (based on preliminary evaluation phase trajectory data) and nominal third stage performance is believed to be due primarily to the excessive velocity and altitude deviations at MECO. These non-nominal conditions necessitated BTL to guide for the larger than nominal orbit. The deviations in the achieved orbit were due to a combination of the greater than nominal third stage performance (discussed previously) and the body attitude errors in pitch and yaw at injection.

The velocity error was due to a higher than nominal engine performance coupled with a third stage inert weight discrepancy. The third stage pre-flight weight breakdown transmitted to BTL for use in their orbit calculations was 2.48 pounds greater in inert weight than the final preflight weights. This resulted in a velocity error contribution at injection of +35 feet per second. This deviation coupled with a velocity error contribution of +51 feet per second due to greater than nominal third stage engine performance resulted in a total velocity deviation at injection of +86 feet per second.

The effective pitch and yaw attitude errors were -1.23 degrees (nose-down) and +0.204 degrees (nose-right), respectively. These values are well within the three-sigma values of 3.52 degrees in pitch attitude and 3.54 degrees in yaw attitude determined for this flight.

The achieved orbit apogee and perigee altitudes are within the three-sigma dispersions of +78, -1 nautical miles for apogee and +2, -47 nautical miles for perigee, predicted for the Tiros D mission.

3.2 Subsystem Performance

3.2.1 Propulsion System

3.2.1.1 First Stage

Over-All Propulsion System

The main engine start sequence main stage operation, and cutoff sequences were normal. Vehicle liftoff occurred 3.59 seconds after engine start command. Total missile thrust, mixture ratio, specific impulse and propulsion system propellant flowrate at 10 seconds were 153,300 pounds, 2.24, 248.3 seconds, and 617.8 pounds per second, respectively. Total missile thrust, mixture ratio, specific impulse, and propulsion system propellant flowrate at main engine cutoff were 175,200 pounds, 1.94, 288.8 seconds, and 604.4 pounds per second, respectively.

The actual flight propellant flowrate data were obtained by adjusting the predicted (computer) flowrates based on the actual propellants loaded, the residual propellants at main engine cutoff, and the actual main engine operation time.

The propulsion sequence of events was normal. The times for fuel float switch actuation liquid oxygen float switch actuation, main engine cutoff, and vernier engine propellant exhaustion were 147.4, 151.6, 159.3 and 172.3 seconds respectively.

Main engine shutdown was commanded by a series-parallel network of four fuel injector pressure switches as a result of fuel depletion. Fuel depletion was verified by the fuel pump inlet pressure and main fuel tank differential pressure measurements (figure 10) as well as fuel float switch data. The liquid oxygen residual at main engine cutoff was 50 pounds. This resulted in a propellant utilization of 99.95%.

Main Engine

Main engine performance was normal throughout the flight. This was indicated by main engine chamber pressure, turbopump speed, propellant pump inlet pressures and turbine inlet temperature data. The main engine burned for 159.3 seconds, which was 0.3 seconds less than predicted nominal. Comparisons of flight thrust, mixture ratio, propellant flowrate, and specific impulse are presented in figures 11 through 14. The actual impulse was determined to be within 0.2% of the 26.96×10^6 lb-sec nominal value.

Vernier Engines

The vernier engines appeared to operate normally throughout flight in spite of some erroneous vernier chamber pressure data. Vernier engine No. 1 chamber pressure data indicated a thrust of 1,040 pounds per engine at 10 seconds, 1,120 pounds per engine at main engine cutoff, and 1,095 pounds per engine during solo operation--- which were almost nominal, but were not considered to be entirely accurate for MECO and solo conditions.

It was noted that the decay of the vernier engine chamber pressure data at main engine cutoff was not characteristic of previous Delta flights, (see figure 15). This was believed to have been caused by a restriction in the line between the chamber and transducer, or in the transducer itself, (see figure 16). This caused the vernier engine chamber pressure data during solo conditions to indicate a value only 10 psi less than that measured during pump-fed operation, until the differential pressure was great enough to expel the restriction. This did not appear to affect the actual function of the vernier engine but did give a false reading from approximately 100 seconds through vernier engine propellant exhaustion. The vernier engine solo duration was approximately 13.0 seconds.

The propellant supply and pressurization system performance was satisfactory throughout flight.

3.2.1.2 Second Stage

The operation of the second stage propulsion system was satisfactory throughout the powered flight time. Propulsion system operating duration was 108.3 seconds as compared to a nominal burning time of 108.9 seconds. Engine cutoff was commanded by BTL guidance at the desired velocity During second stage operation, the average thrust was 7500 lbs, as determined from telemetered chamber pressure data and computed trajectory reconstruction.

The average specific impulse for the flight was 265 seconds. The total average flow rate was 28.3 pounds per second and the average mixture ration was 2.77. The propellant consumption at command cutoff was 95%. If the engine had been allowed to operate to oxidizer depletion (an additional 5 sec), the propellant utilization would have been 99.3 per cent. This data was determined from a computer reconstruction of the flight results.

The initial helium tank pressure of 1605 psia at second stage ignition decayed gradually to 380 psia at SECO. The telemetered helium pressure data showed that the heat generator (HGA) ignited at 173 seconds, indicating normal operation.

Cutoff impulse, determined from telemetered chamber pressure data was 2650 lbs-sec. Cutoff impulse was evaluated from command cutoff signal to zero engine chamber pressure. Predicted cutoff impulse for this flight was 2400 $\pm$ 300 lb-sec.

The helium pressure switch (HPS_2) was monitored again for this flight. Prior to liftoff the switch cycled at a rate of approximately 0.035 cps as compared to a calculated normal cycling rate of 0.03 to 0.05 cps. Approximately nine cycles of the HPS_2 were observed during first stage powered flight with rapid cycling occurring during the transition through Mach 1 and maximum Q, thus confirming previous observations that the switch is vibration sensitive.

The externally mounted gaseous-nitrogen retro system was employed on the second stage to provide the required second to third stage separation distance prior to third stage ignition. The summation pressure of the retro bottles at liftoff was 4550 psia. The pressure reached a maximum of 5470 psi at 213 seconds as a result of aerodynamic heating and then slowly decreased to 5340 psi at second/third stage separation. Telemetered data indicated satisfactory operation of the retro system. The curve in figure 17 is a plot of calculated separation distance versus time based on telemetered data. This plot indicates that there was a separation distance between the second and third stage of 78 feet with a nominal third stage ignition delay of 13.5 seconds after separation.

3.2.1.3 Third Stage

Third Stage motor (SV-117) performance was satisfactory as determined from second stage telemetry and the achieved orbit.

Third stage/payload spinup occurred at 705.0 seconds. Telemetered data indicated that a spin rate of 116 rpm was attained. The pre-flight predicted nominal spin rate was 125 rpm $\pm$ 10% based on a roll moment of inertia of the total spinning mass of 18.73 slug-ft^2 and an efficiency factor for this spin rocket configuration of 80.5 per cent. Results of a post flight analysis of this flight and the previous six Delta flights indicated the efficiency factor for the Tiros D flight should have been 75 percent.

Third stage separation from the second stage occurred at 707.0 seconds. Since the third stage vehicle does not carry a telemetry system third stage performance and burning duration were not monitored. Therefore, third stage ignition was assumed to have occurred at the programmed time of 13.5 seconds after third stage separation and the burning duration was 42 seconds. The achieved orbit of the Tiros D satellite and the vehicle's spatial condition at third stage separation from the second stage indicate performance of the propulsion system was within design tolerances. The predicted total impulse for motor SV-117, based on the manufacturer's propellant weight and nominal specific impulse was 116,060 pound-seconds.

3.2.2 Flight Control System

3.2.2.1 First Stage

The first stage control system of the vehicle performed satisfactorily throughout flight.

Liftoff transients were small and normally damped. Thrust misalignments at liftoff were -0.10 degree in pitch and 0.00 degree in yaw. Thrust misalignments at MECO were +.25 degree in pitch and -.08 degree in yaw.

Maximum main engine deflection in the high wind shear region were +1.09 degree and -1.09 degree in yaw.

Attitude errors at separation were -0.19 degree in pitch, 0.00 degree in yaw, and -0.20 degree in roll.

Rates at separation were 0.19 degree/second in pitch, -0.05 degree/second in yaw, and -0.07 degree/second in roll.

In the transonic region (48-50 seconds) this vehicle experienced extraordinary external forces, inducing maximum pitch, yaw, and roll attitude errors of +0.14, +0.71 and +0.82 degrees respectively.

In the early stages of post-flight analysis, the loss of a fin was investigated as the possible cause of the appearance of external forces. However, film data revealed that all fins were intact up to 63 seconds of flight. Data of previous Delta flights were then surveyed, revealing that past flights have experienced these forces through the same region.

3.2.2.2 Second Stage

Separation and ignition transients in pitch and roll were normal, being well damped out in 10 seconds. Liftoff transients in the yaw plane indicate oscillation (0.4 degrees peak to peak attitude error) for 29 seconds before being damped out. Resulting calculations show the yaw attitude gain ratio to be 0.91 in value. The nominal gain ratio is 2.022 deg/deg. A similar analysis of the pitch gains showed them to be nominal. The mission was not degraded due to these oscillations, however, further investigations are being made to determine the cause of the low gains and to find a method of preventing similar occurrences in the future.

A sustained counterclockwise roll moment is present. This torque has been noted on previous flights and attributed to a swirling motion of the flame as it leaves the nozzle and passes through the motor bell. However, on this flight, an additional counterclockwise torque is impulsively applied at the same time that "jettison fairing" occurs (183.5 seconds). The exact cause of this additional torque is unknown at this time. With these external moments, the roll control gas jets do maintain the vehicle roll attitude error within the dead band of $\pm$ 1.25 degrees.

The programmer of each stage functioned well within performance tolerances.

3.2.3 Vehicle Electronics System

The Delta vehicle employed continual second stage programmer operation from MECO to third stage ignition. A comparison of actual and nominal programmer commands is shown in the following table:

PROGRAMMER COMMAND	TIME (SECONDS) NOMINAL	ACTUAL
MECO (programmer start)	159.5	159.2
Second Stage Ignition	163.5	163.7
Start Powered Flight Pitch Program	165.5	165.4
Nose Fairing Jettison	183.5	184.0
Stop Powered Flight Pitch Program	260.4	260.4
Start Coast Phase Pitch Program	300.5	300.4
Stop Coast Phase Pitch Program	358.0	357.1
Fire Spin Rockets and Initiate Third Stage Time Delay Relay	705.0	705.0

As indicated in the table, programmer command times were nearly nominal. The time interval between programmer start at MECO and third stage ignition was 560.9 seconds compared to a nominal interval of 561.0 seconds.

3.2.4 Mechanical System

Analysis of vehicle performance is based upon the telemetry data received from the vehicle during flight and ground monitoring before flight.

Hydraulic system of the first stage functioned satisfactorily and within normal performance tolerances during flight. There were no detrimental effects to the normal operation of the flight control system because of the frozen ground hydraulic supply line.

Normal second stage separation was achieved by satisfactory operation of the first to second stage and blast band explosive bolts.

Second stage hydraulic control system operated satisfactorily throughout the powered flight of the second stage.

Satisfactory performance of the payload fairing explosive bolts and separation spring cartridges was indicated by the jettison of the payload fairing. At fairing separation (184.0 seconds) a disturbance in the roll axis only was noted. There was no indication of a disturbance in the yaw or pitch axis during fairing separation.

Normal operation of the second to third stage explosive bolts and attach clamp and the third stage igniter wire cutters provided satisfactory third stage separation.

Payload separation and tumble system activation were satisfactory based upon results achieved by the payload.

Tiros D like Tiros A3, was equipped with the combination third stage payload shield and light diffuser. Again there was excellent protection to the camera lenses and infra-red sensors as indicated by the high quality data received from the payload.

TABLE I

SEQUENCE OF EVENTS

Event	Time After Liftoff (Sec.) Nominal	Actual
Start first stage roll program	2	2.11
Stop first stage roll program	9	9.21
Start four-step pitch program	10	10.20
Enable stage I closed loop guidance	80	80.00
Start stage I closed loop guidance	90	88.91
Stop pitch program	140	140.21
Stop stage I guidance	153	151.90
Main engine cutoff	159.50	159.21
Second stage ignition signal	163.50	163.71
Start second stage pitch program	165.50	165.41
Start closed loop guidance	175.90	175.31
Jettison nose fairing	183.50	184.00
Stop pitch program	260.44	260.61
Stop closed loop guidance	266.50	263.16
Start open loop steering	266.90	263.86
Stop open loop steering	268.70	265.81
Engine cutoff command signal	272.10	271.81
Second stage engine cutoff	272.44	272.11
Start coast phase pitch program	300.50	300.41
Stop coast phase pitch program	358.00	357.71
Fire spin rockets	705.00	705.01
Blow separation bolts	707.00	706.80
Third stage ignition	720.50	720.50*
Stage III engine burn-out	762.50	762.50*
Payload separation	1,370.50	1,370.11*

* Based on nominal times.

TABLE II

SUMMARY OF VEHICLE PERFORMANCE PARAMETERS

Lift-Off Conditions	Units	DTO Nominal	Actual
Weight	lbs	111,776	111,639
Thrust (1)	lbs	151,550	152,900
Specific impulse (1)	sec	246.1	248.0
Weight of propellant loaded for impulse	lbs	97,949	97,829
Total Thor payload weight	lbs	5,775	5,777
Guidance Initiation			
Time	sec	90	88.91
Velocity (relative to launch point)	ft/sec	2,795	2,905
Flight path elevation angle (2)	deg	49.64	50.56
Flight path azimuth angle (2)	deg	46.37	49.68
Range	ft	41,419	43,745
Altitude	ft	79,239	80,936
Main Engine Cut-Off			
Time	sec	159.50	159.26
Velocity (relative)	ft/sec	14,138	14,546
Flight path elevation angle (2)	deg	35.39	35.32
Flight path azimuth angle (2)	deg	46.93	46.95
Longitude	deg	79.67	79.58
Geodetic latitude	deg	29.20	29.25
Range	ft	413,795	429,819
Booster 11P time	sec	951.5	990.4
Booster 11P range	n. mi.	1,414	1,514
Altitude	ft	383,019	396,068
Weight	lbs	13,997	13,637
Thrust	lbs	173,578	175,200
Specific impulse	sec	286.6	288.8
Residual propellant	%	0.4	0.05

TABLE II (CONT'D)

Second Stage Ignition	Units	DTO Nominal	Actual
Time	sec	163.50	163.7
Velocity (relative)	ft/sec	14,085	14,500
Flight path elevation angle(2)	deg	35.13	35.18
Flight path azimuth angle(2)	deg	46.99	46.95
Range	ft	459,086	481,593
Altitude	ft	415,605	433,348
Weight(7)	lbs	5,516	5,516
Thrust(7)	lbs	6,796	6,723
Specific impulse(7)	sec	264.5	265
Propellant loaded for impulse	lbs	3,231.5	3,262.0
Nose Fairing Jettison			
Time	sec	183.50	184.0
Weight of fairing	lbs	197.4	199.6
Second Stage Engine Cut-Off			
Time	sec	272.44	272.11
Velocity (relative to launch point)	ft/sec	18,564	18,609
Flight path elevation angle(2)	deg	16.38	16.86
Flight path azimuth angle(2)	deg	49.16	49.31
Longitude	deg	76.02	75.91
Geodetic latitude	deg	32.04	32.09
Range	n. mi.	319.7	325.9
Altitude	n. mi.	200.9	206.7
Weight	lbs	2,209	2,264
Thrust (steady state)	lbs	7,551	7,500
Specific impulse (steady state)	sec	265.3	265
Tail-off impulse	lb-sec	2,400	2,650
Total weight consumed	lbs	3,109	3,052
Propellant consumption	%	96.12	95.0

TABLE II (CONT'D)

Third Stage Ignition	Units	DTO Nominal	Actual
Time	sec	720.50	720.11
Burn time	sec	42.0	42.0
Velocity (relative to launch point)	ft/sec	16,867	16,803
Flight path elevation angle(2)	deg	-1.013	-0.279
Flight path azimuth angle(2)	deg	63.62	63.70
Longitude	deg	55.88	55.89
Geodetic latitude	deg	42.88	42.81
Euler Attitude Angles(3)			
Pitch (Θm)	deg	-24.23	-25.48
Yaw (ψm)	deg	-0.0	-0.628
Roll (Θm)	deg	0.0	0.184
Range	n. mi.	1,476	1,474
Altitude	n. mi.	381.7	400.7
Weight	lbs	822.8(5)	822.8
			[illegible]

Third Stage Burn-Out	Units	DTO Nominal	Actual
Time	sec	762.50	762.11
Inertial velocity	ft/sec	24,631	24,652
Velocity (relative to launch point)	ft/sec	23,511	23,526
Inertial flight path angle	deg	0.001	-0.375
Flight path elevation angle(2)	deg	0.077	-0.320
Inertial azimuth angle	deg	66.68	66.54
Flight path azimuth angle(2)	deg	65.47	65.35
Longitude	deg	53.34	53.38
Geodetic latitude	deg	43.76	43.69
Euler Attitude Angles(3)			
Pitch (Θm)	deg	-22.23	-25.48
Yaw (ψm)	deg	0.0	-0.628
Roll (ϕm)	deg	0.0	0.184
Inertial attitude angles(4)			
Elevation angle (θ'_{ι})	deg	6.36	3.04
Azimuth angle (ψ'_{ι})	deg	65.05	64.49

TABLE II (CONT'D)

Third Stage Burn-Out	Units	DTO Nominal	Actual
Range	n. mi.	1,599	1,596
Altitude	n. mi.	380.0	399.4
Weight	lbs	357.8(5)	357.8
Total impulse	lb/sec	116,060(6)	116.650

Footnotes:

(1) Average value 0 - 10 seconds.

(2) Angles with respect to relative velocity vector.

(3) Euler angles $\theta m, \psi m, \phi m$. Angles specifying the orientation of the vehicle axes (Xm, Ym, Zm) with respect to an inertial reference platform. Order of rotation: pitch, Θm, about Ym (+ turning Zm into Xm); yaw, ψ'm, about Zm (+ turning Xm into Ym); and roll, ϕm, about Xm (+ turning Ym into Zm), degrees.

(4) θ_L Vehicle centerline elevation angle: angle between the vehicle centerline and a plane perpendicular to the radius vector from the center of the earth to the vehicle, positive for vehicle nose pointing away from the earth, degrees.

ψ'_L Vehicle centerline azimuth angle: angle between the local meridian and the projection of the vehicle centerline onto a plane perpendicular to the radius vector from the center of the earth to the vehicle, positive clockwise from true north, degrees.

(5) Value different from DTO. Based on DAC preflight weight breakdown.

(6) Based on nominal performance of third stage unit SV-117.

(7) At 90% thrust.

357.8
285.
642.8

TABLE III

WEIGHTS ANALYSIS

DELTA BOOSTER S/N 317

	WEIGHT AT LIFTOFF (LBS)			WEIGHT AT MECO (LBS)		
	DTO*	Predicted*	Actual	DTO*	Predicted*	Actual
Dry Booster	6,899	6,882	6,882	6,899	6,882	6,882
Payload	5,775	5,766	5,777	5,775	5,766	5,777
Oxidizer:						
Main Tank	67,790	67,790	67,712	0	0	50
Piping & Engine	160	160	160	160	160	160
Vernier Tank	54	54	45	61	61	61
Fuel:						
Main Tank	30,632	30,632	30,602	394	394	0
Piping & Engine	216	216	216	216	216	216
Vernier Tank	28	28	24	40	40	40
Nitrogen:						
Prop. Tanks & Pneu System	94	94	94	94	94	94
Gaseous Oxygen	0	0	0	311	311	310
Lube Oil	127	127	127	47	47	47
TOTAL VEHICLE	111,775	111,749	111,639 ± 250	13,997	13,971	13,637 ± 2[illegible]

Densitites: Loading; Oxidizer, 71.310 Lb/Ft^3 Fuel, 50.412 Lb/Ft^3

Liftoff; Oxidizer, 70.928 Lb/Ft^3

Estimated at MECO; Oxidizer, 68.848 Lb/Ft^3

	CENTER OF GRAVITY (DACo STA)			MOMENT OF INERTIA ($SLUG-FT^2$)		
	X	Y	Z	PITCH	ROLL	YAW
Liftoff	99.93	398.2	100.07	746,970	3,170.3	746,964
MECO	99.59	247.5	100.63	364,533	2,015.5	364,525

TOTAL AVERAGE RATE OF CHANGE OF BOOSTER WEIGHT = 615.20 Lb/Sec

* Predicted data are based on the latest preflight weights and nominal densities. DTO and predicted weights are based on a 99.6% propellant utilization.

TABLE III (Cont'd)

WEIGHT ANALYSIS

DELTA STAGE II

	WEIGHT AT STAGE II LIFTOFF (LBS)			WEIGHT AT CUTOFF SIGNAL (LBS)		
	DTO*	Predicted*	Actual	DTO*	Predicted*	Actual
Separable Payload	287.00	285.88	285.88	287.00	285.88	285.88
Stage III	537.91	537.84	537.84	537.91	537.84	537.84
Dry Stage II (w/spin prop)	1,189.21	1,117.46	1,177.46	1,189.21	1,177.46	1,177.46
Fuel in Tank	876.65	876.65	876	46.43	42.51	74
Oxidizer in Tank	2,364.82	2,364.82	2,373	88.93	76.10	126
Fuel Trapped in Tank and lines	11.20	11.20	14.22	11.20	11.20	14.22
Oxidizer Trapped in Tank, Lines and TCA	31.00	31.00	30.92	31.00	31.00	30.92
Helium & Grain	15.42	15.42	15.42	12.42	12.42	12.42
Retro System GN_2	5.00	5.00	5.60	5.00	5.00	5.60
Payload Fairing	197.40	199.56	199.56	0.00	0.00	0.00
TOTAL	5,515.61	5,504.83	5,516. ± 10	2,209.10	2,179.41	2,264 ± 10
Starting Propellant	8.06	8.06	8.06			
Transition Skirt	251.32	253.36	253.36			
TOTAL BOOSTER PAYLOAD	5,774.99	5,766.25	5,777. ± 10			
Stop Losses				-26.00	-26.00	-10.19
TOTAL VEHICLE AT STOP LOSSES COMPLETE				2,183.10	2,153.41	2,254. ± 10

Loading Densities: Oxidizer; 93.83192 Lb/Ft^3 Fuel; 49.14257 Lb/Ft^3

Loading Temperatures: Oxidizer; 69.0^{o}F Fuel; 70.0^{o}F

	CENTER OF GRAVITY (DACo STA)			MOMENT OF INERTIA (SLUG-FT^2)		
	X	Y	Z	PITCH	ROLL	YAW
Total Booster Payload	100.1	-114.3	100.0	10,349	93.5	10,346
Stage II at Liftoff	100.1	-120.6	100.0	9,284	65.8	9,283
Vehicle at Cutoff Signal	100.2	-167.1	100.0	5,372	49.7	5,370

* DTO and predicted weights are based on a 97.0% propellant utilization. Predicted weights reflect the latest preflight weights and densities.

APPENDIX I

SUMMARY AND EVALUATION COUNTDOWN

T-1 day countdown began at 0730 EST, T-495 minutes, on February 5, 1962.

Engine checks were completed at 0915 EST and payload checks were run from 0945 EST to 1036 EST. Electrical system checks were started at 0925 EST. No gas discharge was noted from the counter-clockwise (CCW) roll jet during second stage slew checks, or from the pitch down jet when the system was in the coast phase. Gas pressure in the fuel tank was not sufficient to give an appreciable flow; however, the solenoids were operating properly.

On the second stage programmer run using external power, BTL failed to send the SECO command. Another external run was made during which BTL sent a manual SECO.

During second stage slew checks, it was noted on telemetry that the second stage engine pitch position had approximately a 2 percent jump in level near the full down position. Subsequent checks proved that the engine was not jumping at the time of this abnormality and the indication was from an intermittently faulty telemetry potentiometer. As this indication occurred only near the full pitch down position, it was decided to launch with the condition, since only telemetry was affected and operation of the control system would not be jeopardized.

Second stage propellant servicing began at 1400 EST and was completed at 1529 EST. Retro bottle pressurizing started at 1600 EST and was completed at 1646 EST. This concluded the T-1 day countdown.

The first attempt of the flight countdown began at 2203 EST, T-480 minutes, on February 5, 1962.

The countdown proceeded normally with no difficulties encountered until the one hour built-in hold at 0528 EST February 6, 1962. During this hold, the vehicle fairing was removed to make a special check of the spacecraft IR recorder. The tests revealed the recorder to be operating properly.

The terminal countdown began at 0628 EST, T-35 minutes. A hold was called at 0633 EST, T-30 minutes, because the pad had not been cleared. The hold was released at 0644 EST. After the missile bottle pressure was increased to 3000 psi, it was noted that the engine pneumatic regulator read approximately 50 psi higher than nominal. The liquid oxygen tank vent valve was cycled several times to provide gas flow through the regulator. As this did not lower the pressure within specification, a hold was called at 0705 EST, T-9 minutes. An inspection crew went to the vehicle to correct the difficulty. The regulator was reset and checked out by pressurizing and depressurizing the start tanks.

Upon completion of the inspection and regulator setting, the launcher work platform under the engine section was found to be inoperative and had to be moved into the retired position with hand operated equipment. It was then determined that insufficient time remained to permit launch during the permissible launch time interval. It was therefore necessary to reschedule the launch for February 7.

The second attempt of the flight countdown began at 2333 EST, T-390 minutes, on February 6, 1962.

The countdown proceeded without incident until T-2 minutes when it was noted that the first stage roll rate gyro spin motor monitor was exhibiting an erratic starting trace on the blockhouse recorder. An effort to establish that the difficulty was in the ground electronics equipment was unsuccessful. As insufficient time than remained to correct the difficulty within the permissible launch time interval, the launch was rescheduled for February 8. Later it was found that the erratic trace was caused by a bad amplifier in the blockhouse monitor.

The successful flight countdown began at 0103 EST, T-300 minutes on February 8.

The countdown proceeded normally to 0538 EST when a one hour built-in hold was entered.

The terminal countdown began at 0628 EST, T-35 minutes, and proceeded to T-12 minutes at which time a hold was entered because of a lack of hydraulic pressure in the first stage. The count was recycled to T-15 minutes and a crew was sent to the launcher. It was found that the vernier engine No. 1 liquid oxygen bleed was chilling the hydraulic supply line in the launch ring. The line was thawed with a stream of water and hydraulic flow was attained. During the remainder of the countdown, the engine was slewed several times to maintain hydraulic flow through the line.

The countdown was functionally talked down to T-9 minutes and continued to liftoff which occurred at 0743:45.690 EST.

APPENDIX II

DESCRIPTION OF CONFIGURATION

The Tiros D vehicle consisted of three stages designed and built by Douglas Aircraft Company. The spacecraft was furnished by NASA.

A basic configuration description will be found in reference 3.

Following is a description of the variations from the basic vehicle, described in reference 3, which apply to the Tiros D vehicle.

<u>Second Stage</u>

a. Oxidizer probes were used as a backup cutoff source.

b. The thrust chamber pressure switch was armed at the same time as the oxidizer probes.

c. A gaseous nitrogen retro system was employed.

d. Two 1KS40 and six .6KS40 spin rockets were employed to obtain a spin rate of 126 $\pm$ 10 percent.

<u>Third Stage</u>

a. The third stage motor number was SV-117.

b. A bulbous fairing was employed to protect the payload.

c. A pyrotechnic time delay was attached to the existing igniter paddle to obtain an ignition time delay of approximately 15.5 seconds to delay third stage firing during retro system operation.

d. Aluminum foil was added to the dome of the third stage motor case to prevent emission of contaminates against the spacecraft during third stage burning.

e. The equipment rack had a 650 second spacecraft separation time with a pyrotechnic time delay relay for Yo weight release.

f. A payload shield with light diffusers was used.

The Tiros D Satellite is an 18-sided polygon, 19 inches high and 42 inches in diameter, weighing 285 pounds. Power supply consists of 63 nickel cadmium storage batteries that are rechargeable by 9,260 solar cells. One wide angle camera and one intermediate angle camera provide coverage for viewing cloud systems. A camera lens different from any previously used in Tiros satellites has been installed in one of the camera systems for the purpose of reducing distortion and providing somewhat better resolution in the picture immage while preserving relatively large coverage. From an altitude of 475 nautical miles, this lens will cover an area about 450 nautical miles on a side when the camera is pointing straight downward. The second camera covers an area approximately 750 nautical miles on a side. Two tape recorders are provided. Each recorder will store a sequence of thirty-two pictures for readout when commanded.

Other instrumentation includes remote control electronics, electronic clocks for triggering the cameras when away from the ground stations, beacon transmitters, horizon scanners, telemetry circuits and a magnetic orientation control system. Nearly identical infra-red experiments to those in Tiros II and III also are carried.

Five transmitters relay data from the satellite to the ground stations. Each of the two television camera systems has a two-watt transmitter operating on 235 megacycles. One two-watt 237.8 megacycle transmitter relays infra-red experiments data. Two tracking beacons operating continuously on frequencies of 136.23 megacycles and 136.92 megacycles are used to relay satellite telemetry data.

REFERENCES

1. Preliminary Flight Test Report No. 7 Delta Mission No. 7 Vehicle 317/2020/3020 Tiros D Spacecraft. Model DM-19 (C)

2. Detailed Test Objectives - Delta Launch Vehicle S/N 317/2020/3020 Payload Tiros D, Douglas Report SM-39170, December 1961 (C)

3. Model Specification Delta Space Research Vehicle - Douglas Specification DS-2105A, Issued 28 August 1959 (C)

4. General Test Plan for Delta Space Research Vehicle, Douglas Report SM-35760, Revision A, October 1959 (C)

DELTA TIROS D
ALTITUDE VS RANGE
BOOST PHASE

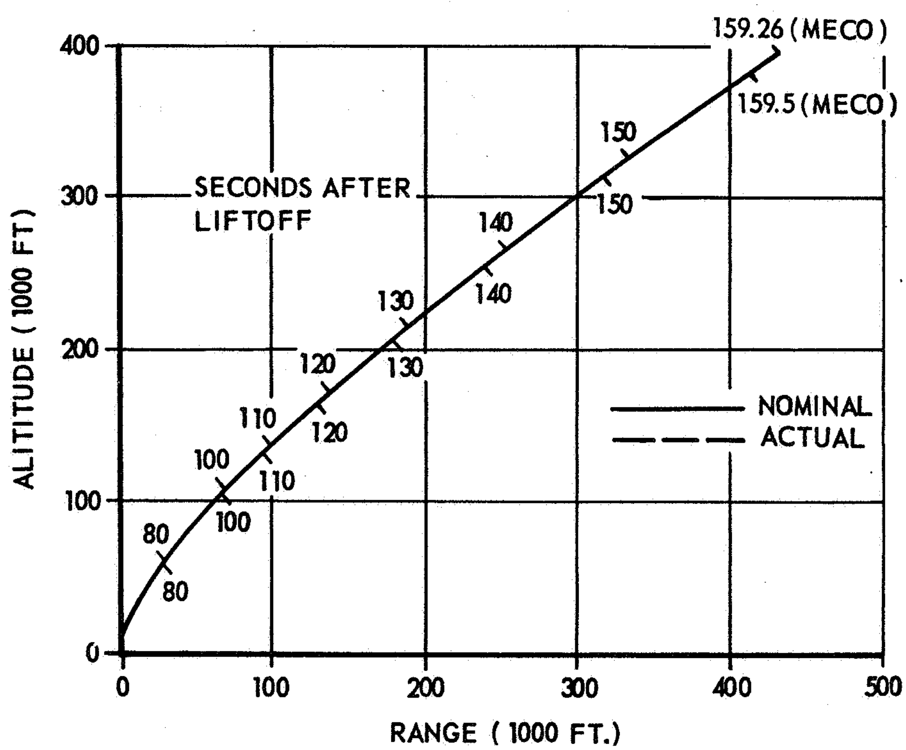

FIGURE 1

DELTA TIROS D
LATITUDE VS LONGITUDE
BOOST PHASE

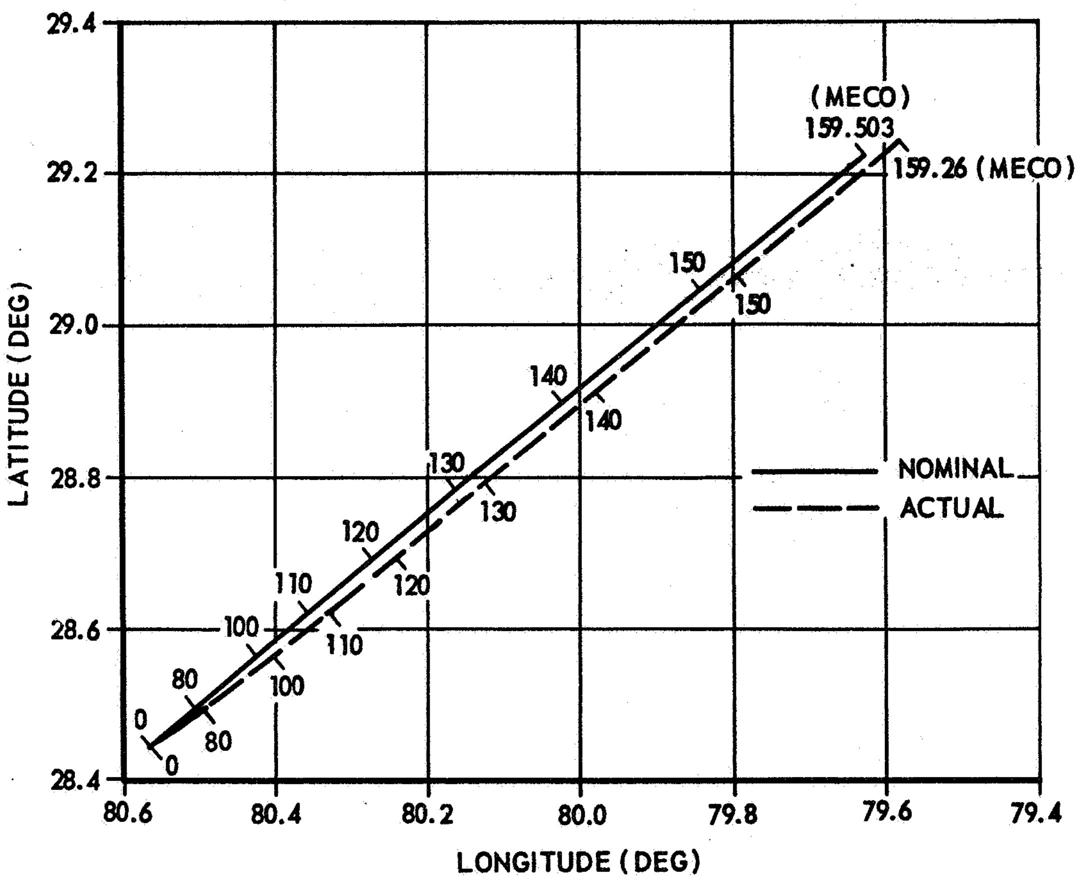

FIGURE 2

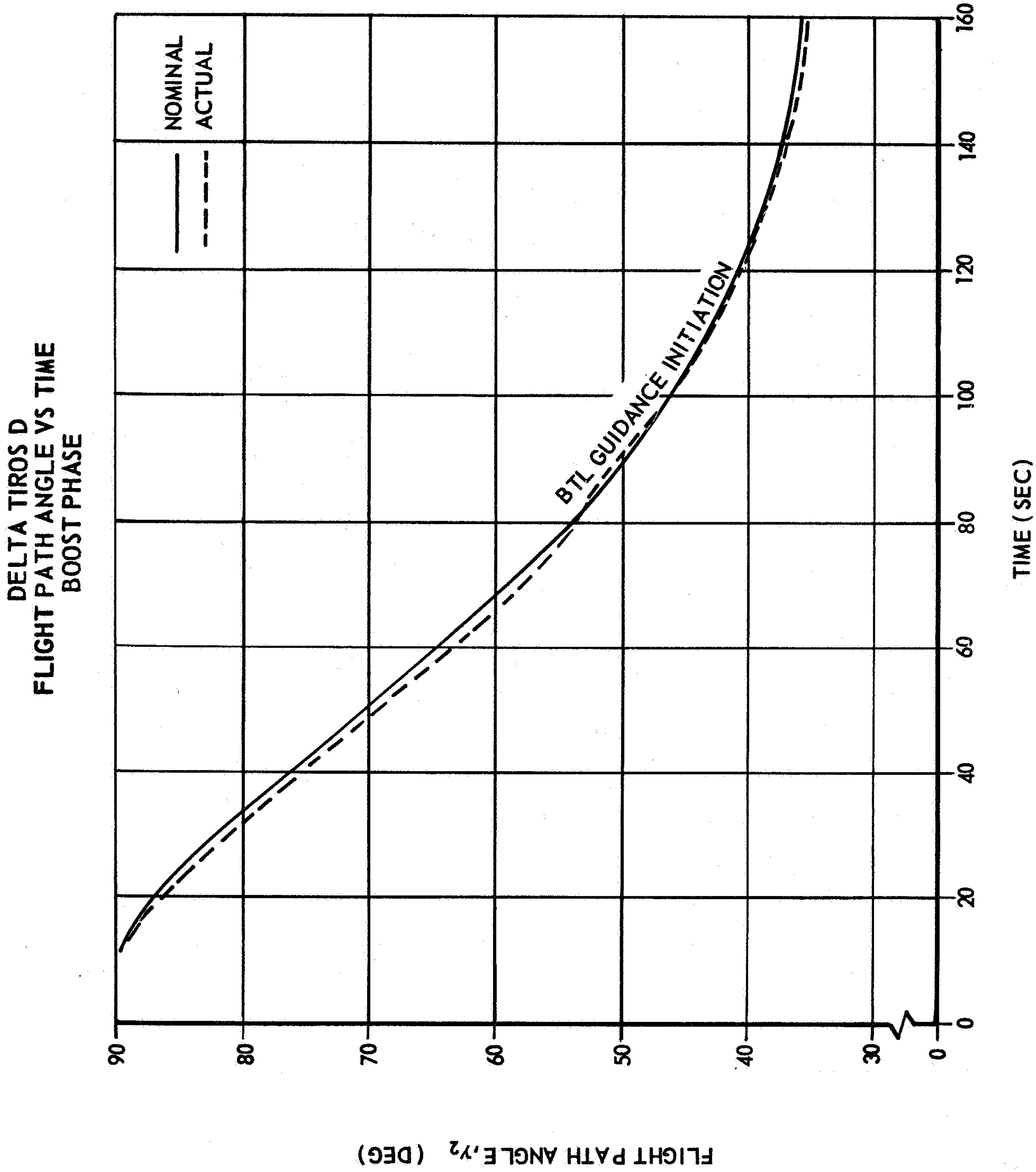
DELTA TIROS D
FLIGHT PATH ANGLE VS TIME
BOOST PHASE
NOMINAL
ACTUAL
BTL GUIDANCE INITIATION
TIME (SEC)
FLIGHT PATH ANGLE, γ2 (DEG)

FIGURE 3

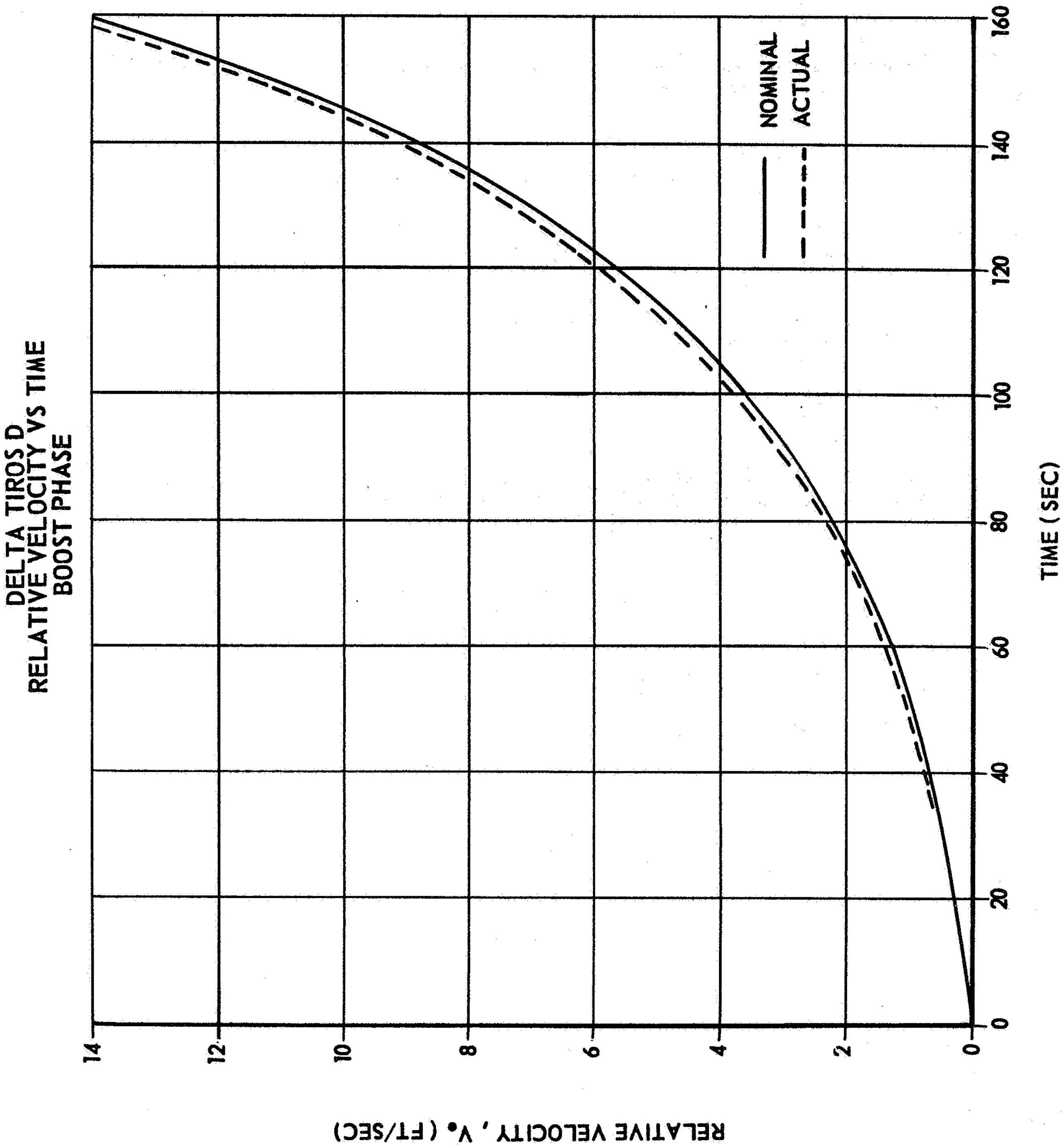
DELTA TIROS D
RELATIVE VELOCITY VS TIME
BOOST PHASE
NOMINAL
ACTUAL
RELATIVE VELOCITY, V_e (FT/SEC)
TIME (SEC)
0
2
4
6
8
10
12
14
0
20
40
60
80
100
120
140
160

FIGURE 4

IIP FOR NOMINAL AND ACTUAL TRAJECTORIES TIROS D

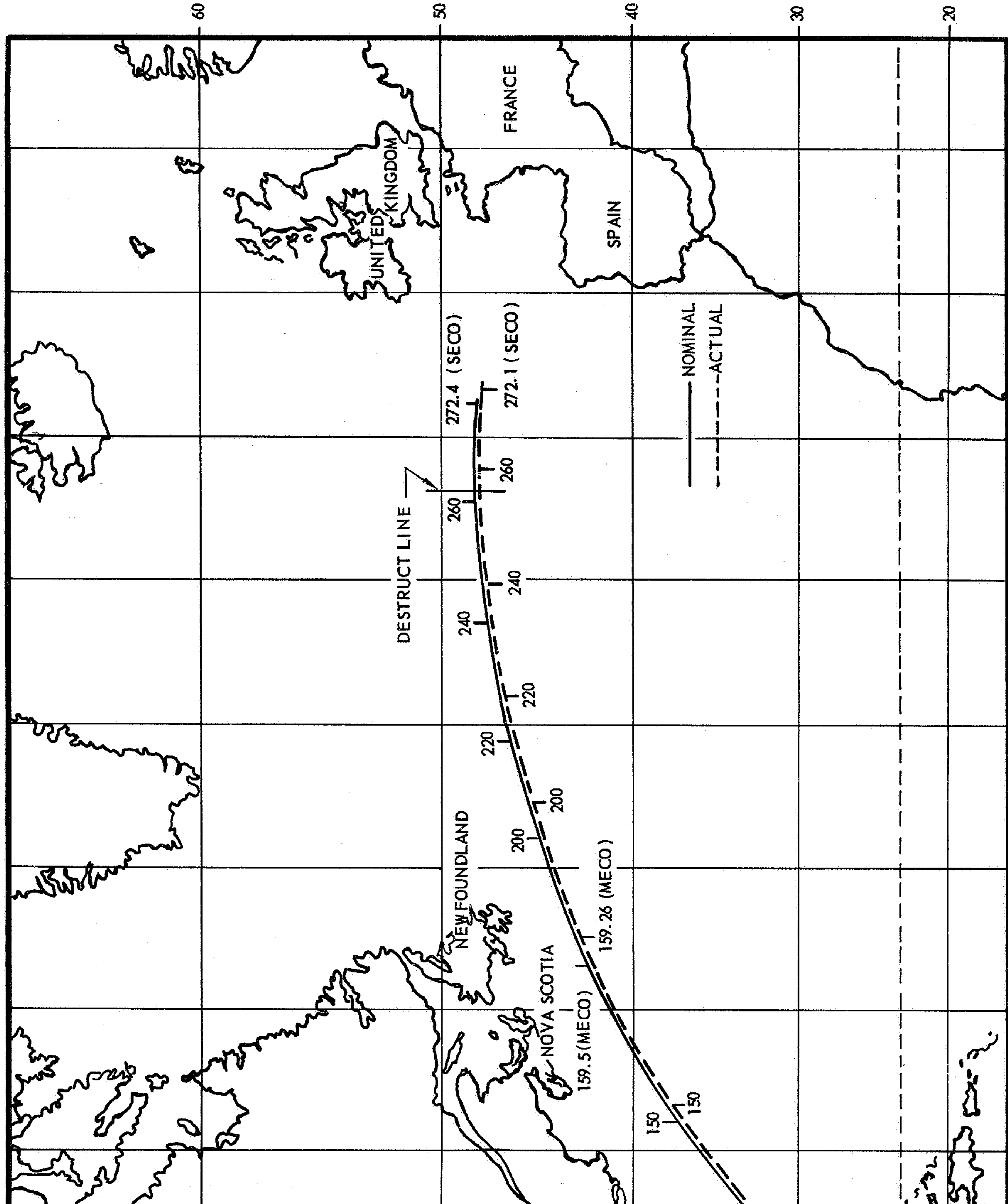

FIGURE 5

DELTA TIROS D
ALTITUDE VS RANGE
SECOND STAGE PHASE

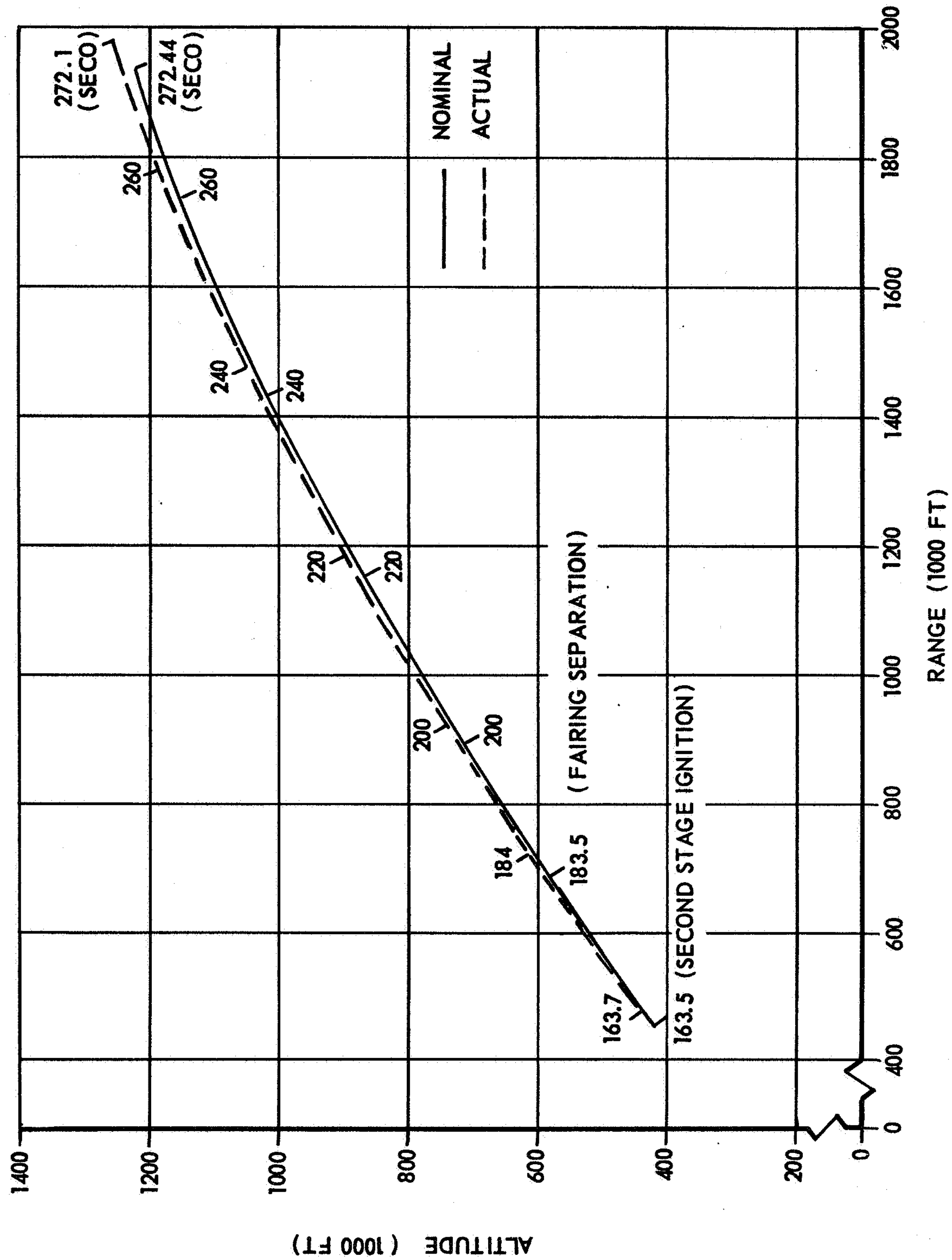

FIGURE 6

DELTA TIROS D
LATITUDE VS LONGITUDE
SECOND STAGE PHASE

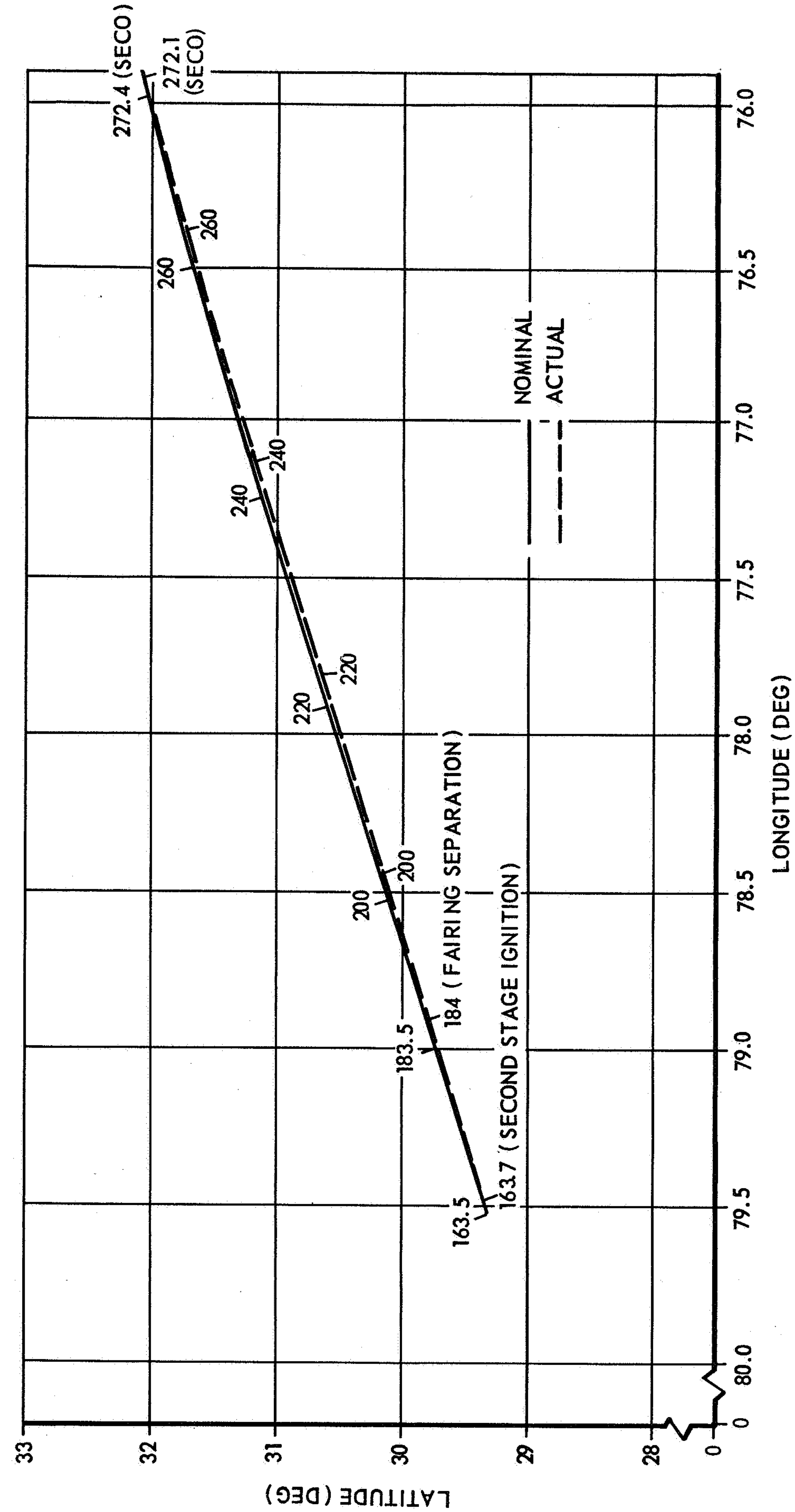

FIGURE 7

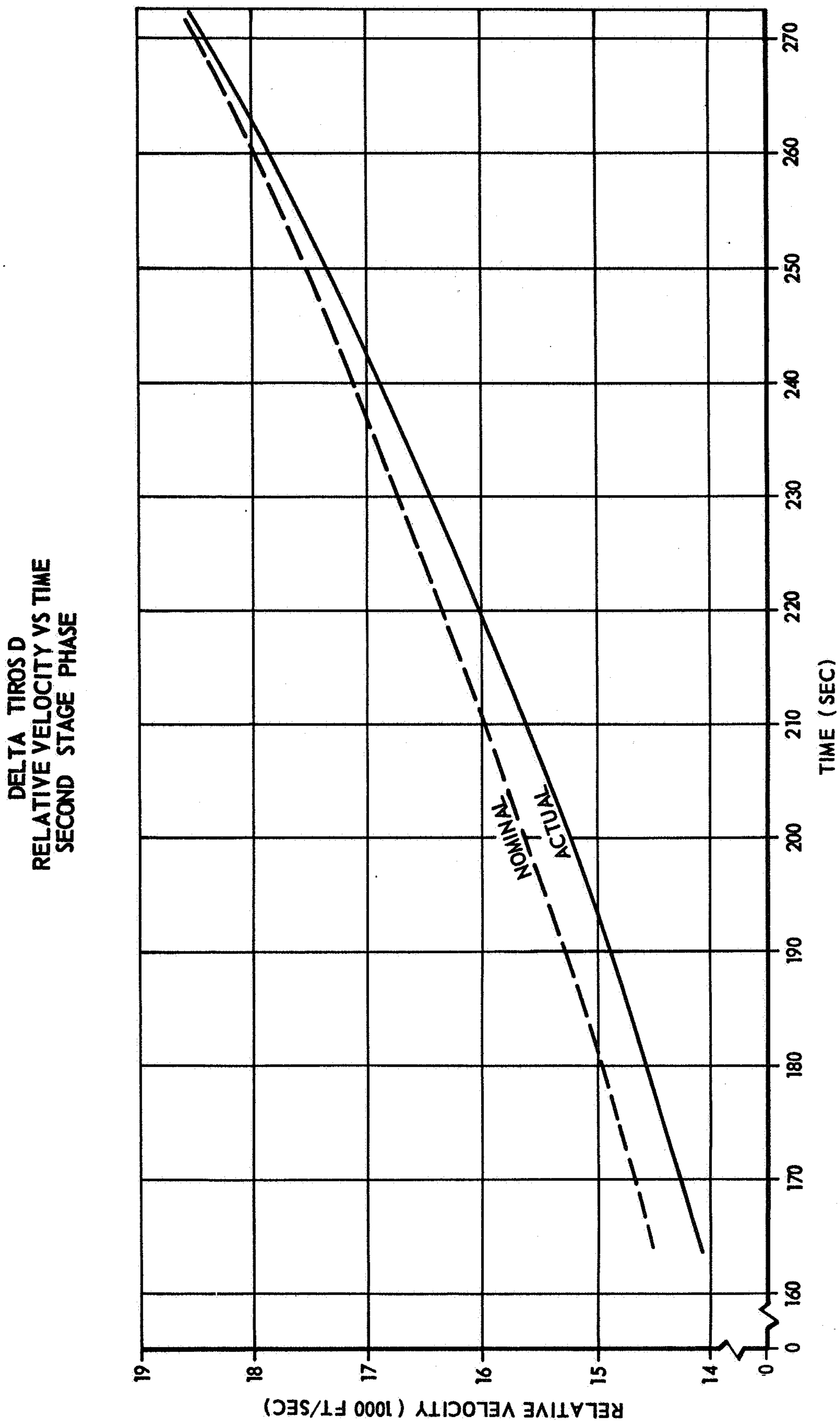
DELTA TIROS D
RELATIVE VELOCITY VS TIME
SECOND STAGE PHASE
NOMINAL
ACTUAL
RELATIVE VELOCITY (1000 FT/SEC)
TIME (SEC)
0
14
15
16
17
18
19
160
170
180
190
200
210
220
230
240
250
260
270

FIGURE 8

DELTA TIROS D
FLIGHT PATH ANGLE VS TIME
SECOND STAGE PHASE

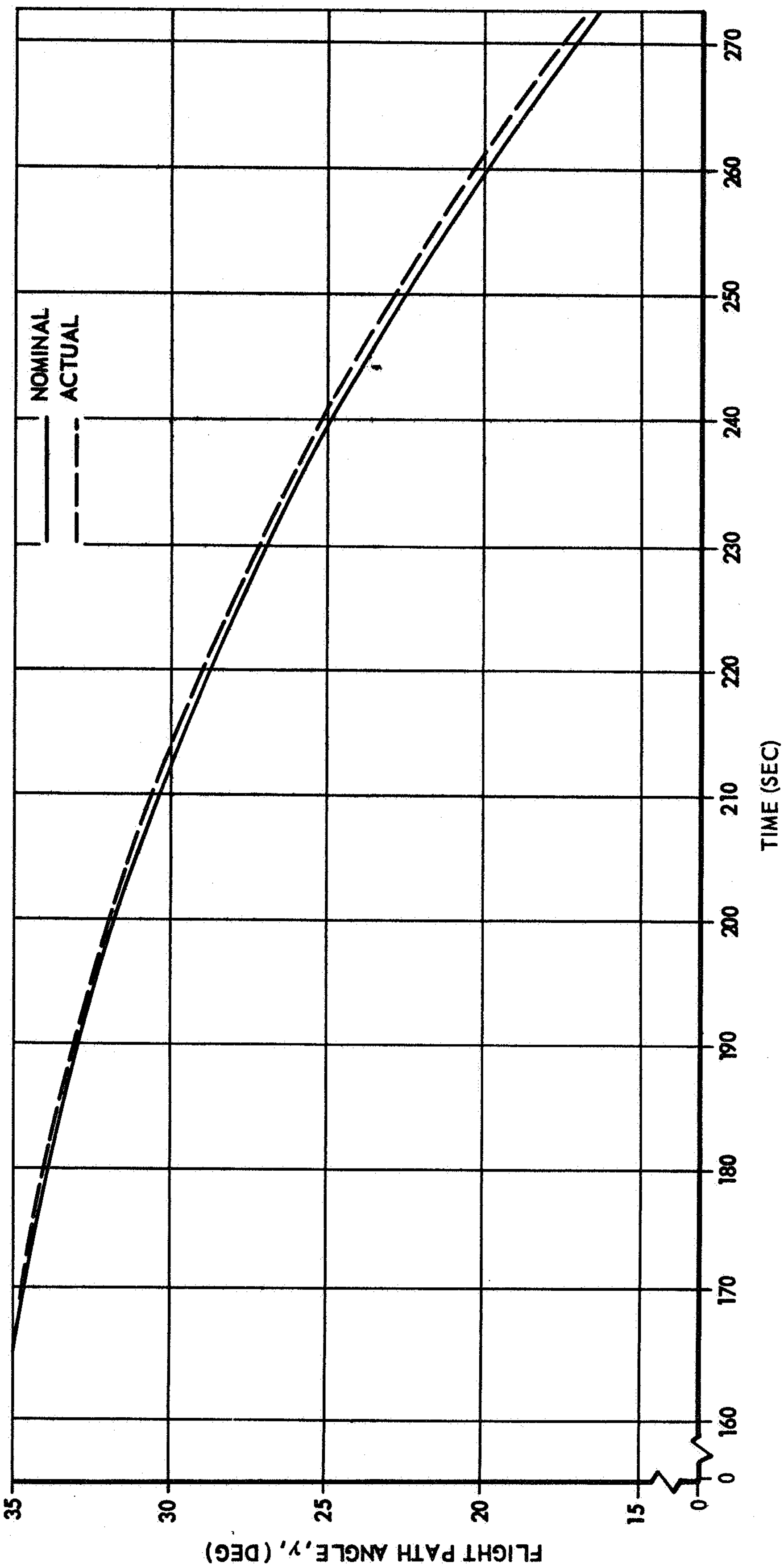

FIGURE 9

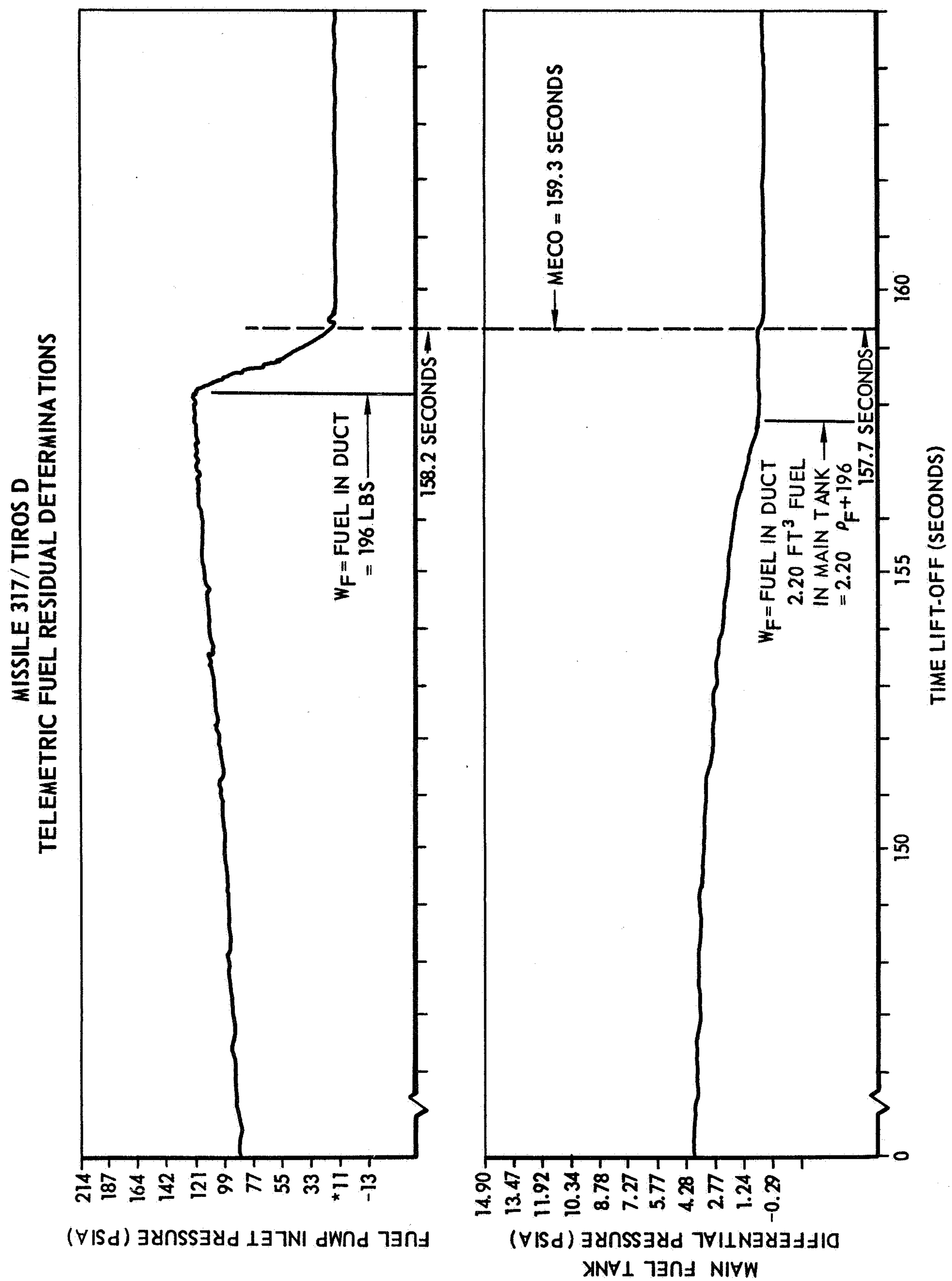
MISSILE 317/TIROS D
TELEMETRIC FUEL RESIDUAL DETERMINATIONS
FUEL PUMP INLET PRESSURE (PSIA)
214
187
164
142
121
99
77
55
33
*11
-13
MAIN FUEL TANK
DIFFERENTIAL PRESSURE (PSIA)
14.90
13.47
11.92
10.34
8.78
7.27
5.77
4.28
2.77
1.24
-0.29
W_F = FUEL IN DUCT
= 196 LBS
158.2 SECONDS
MECO = 159.3 SECONDS
W_F = FUEL IN DUCT
2.20 FT^3 FUEL
IN MAIN TANK
= 2.20 ρ_F + 196
157.7 SECONDS
0
150
155
160
TIME LIFT-OFF (SECONDS)

FIGURE 10

THRUST HISTORY
MISSILE 317/ TIROS D

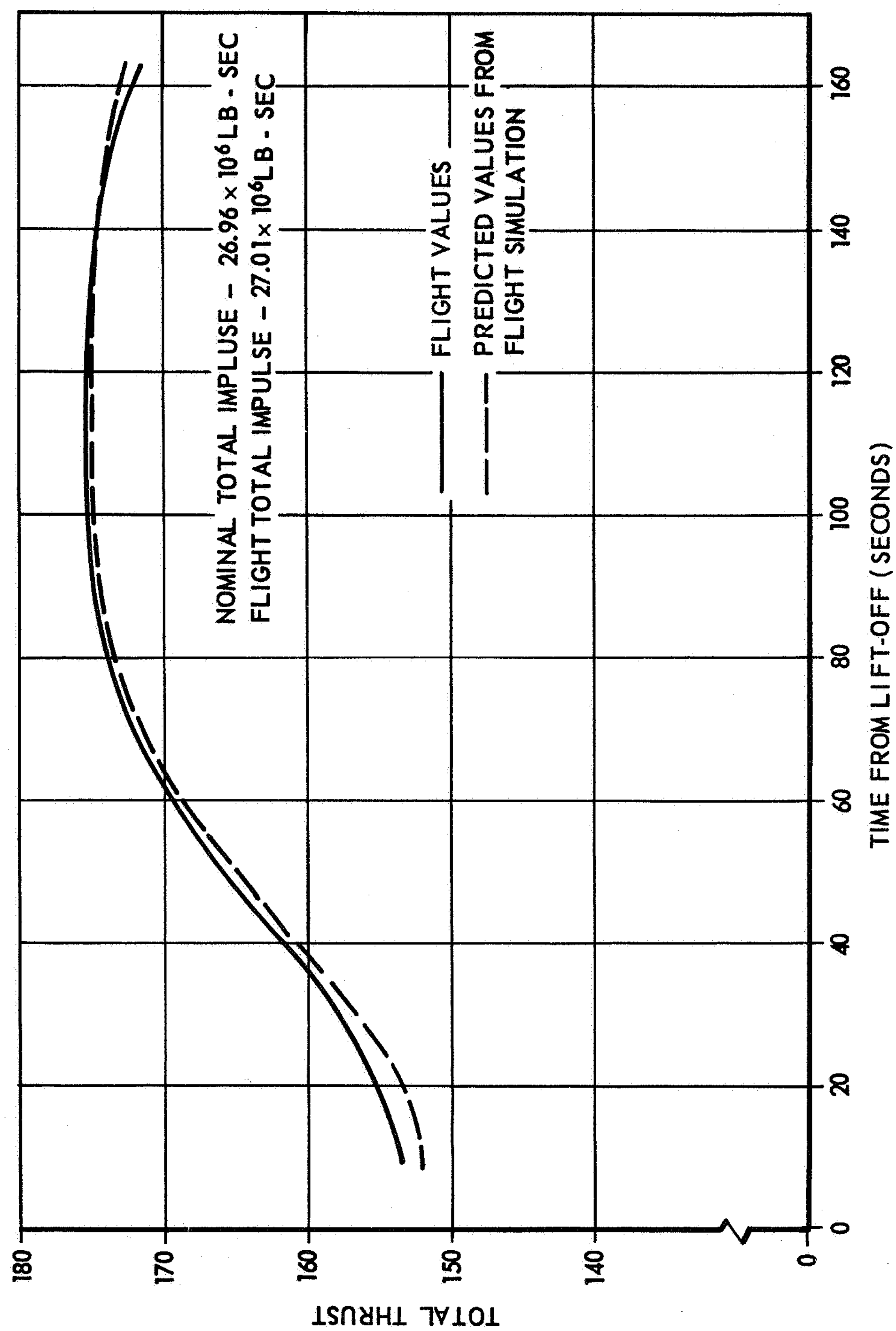

FIGURE 11

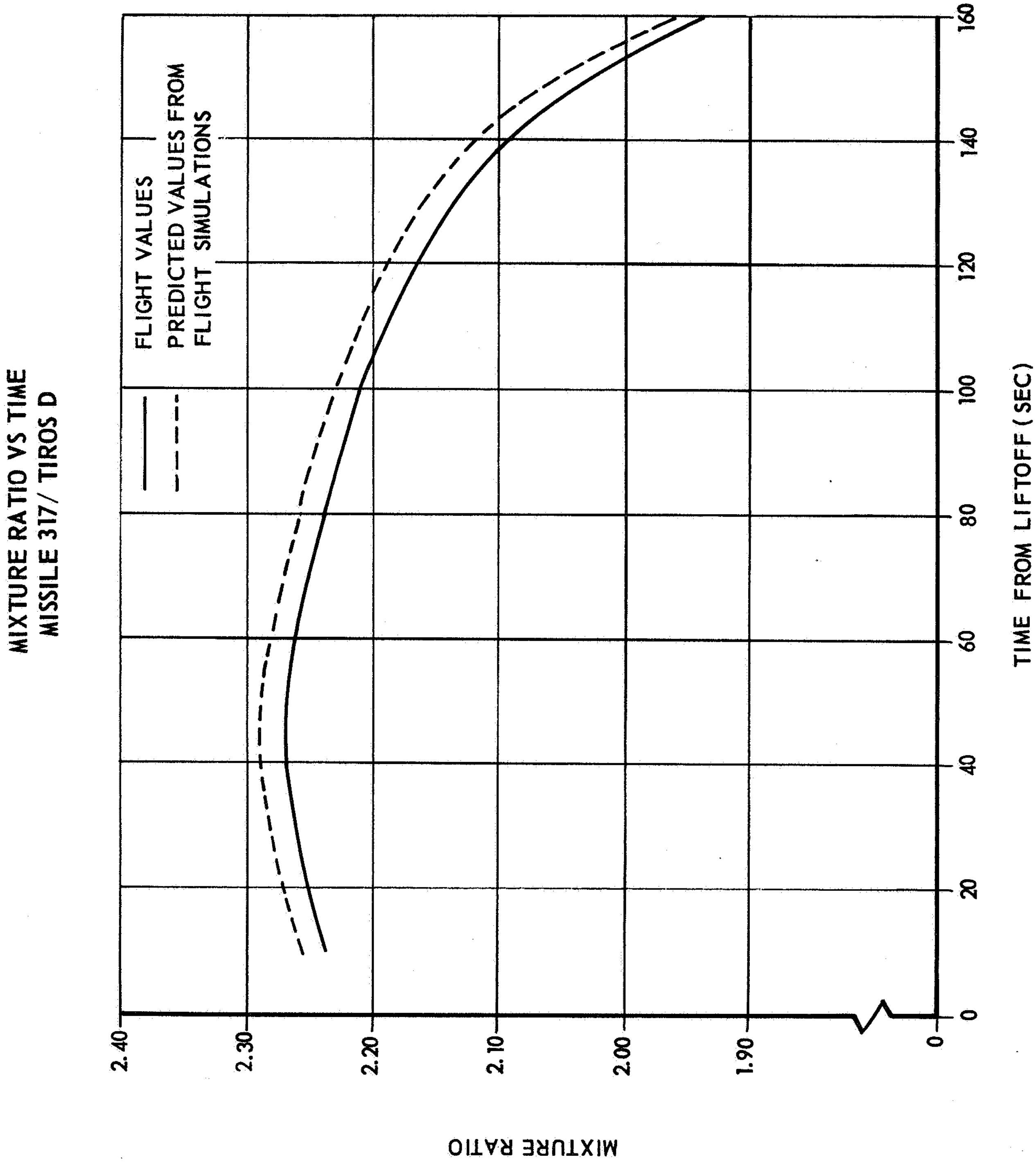
MIXTURE RATIO VS TIME
MISSILE 317/ TIROS D
FLIGHT VALUES
PREDICTED VALUES FROM FLIGHT SIMULATIONS
MIXTURE RATIO
2.40
2.30
2.20
2.10
2.00
1.90
0
TIME FROM LIFTOFF (SEC)
0
20
40
60
80
100
120
140
160

FIGURE 12

PROPELLANT FLOWRATE VS TIME
MISSILE 317/ TIROS - D

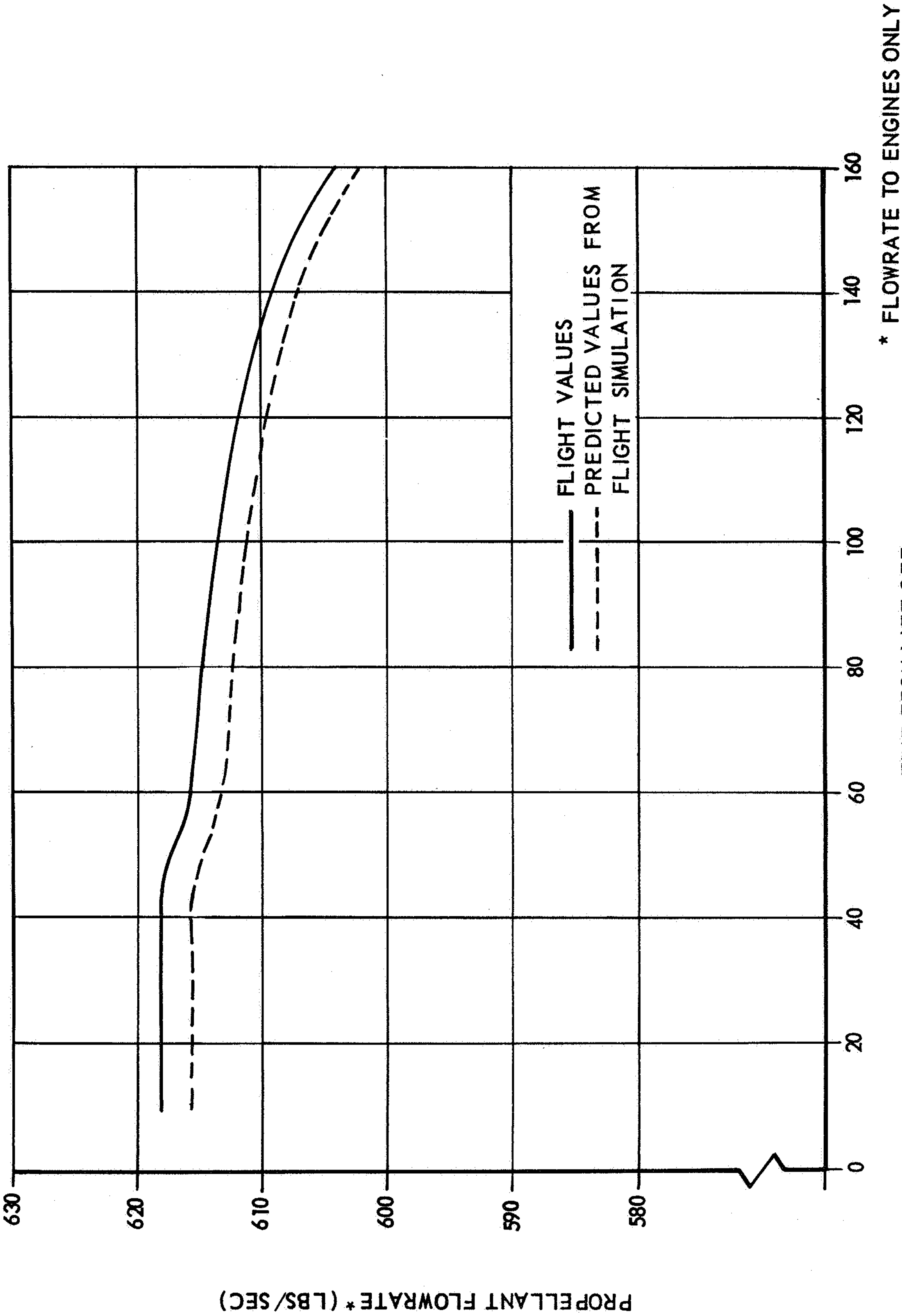

FIGURE 13

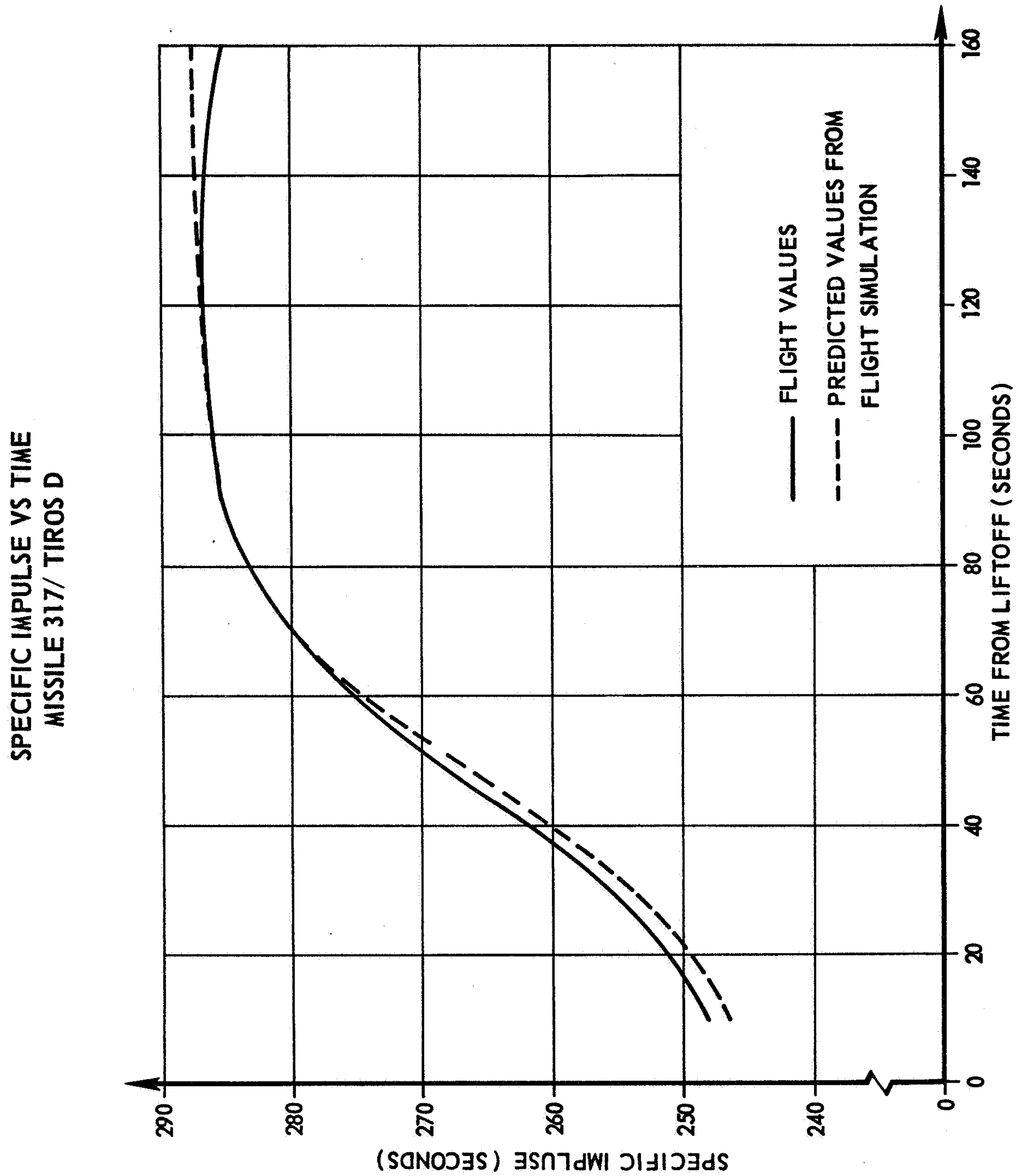
SPECIFIC IMPULSE VS TIME
MISSILE 317/ TIROS D
FLIGHT VALUES
PREDICTED VALUES FROM
FLIGHT SIMULATION
SPECIFIC IMPLUSE (SECONDS)
290
280
270
260
250
240
0
TIME FROM LIFTOFF (SECONDS)
0
20
40
60
80
100
120
140
160

FIGURE 14

VERNIER ENGINE CHAMBER PRESSURE VS TIME

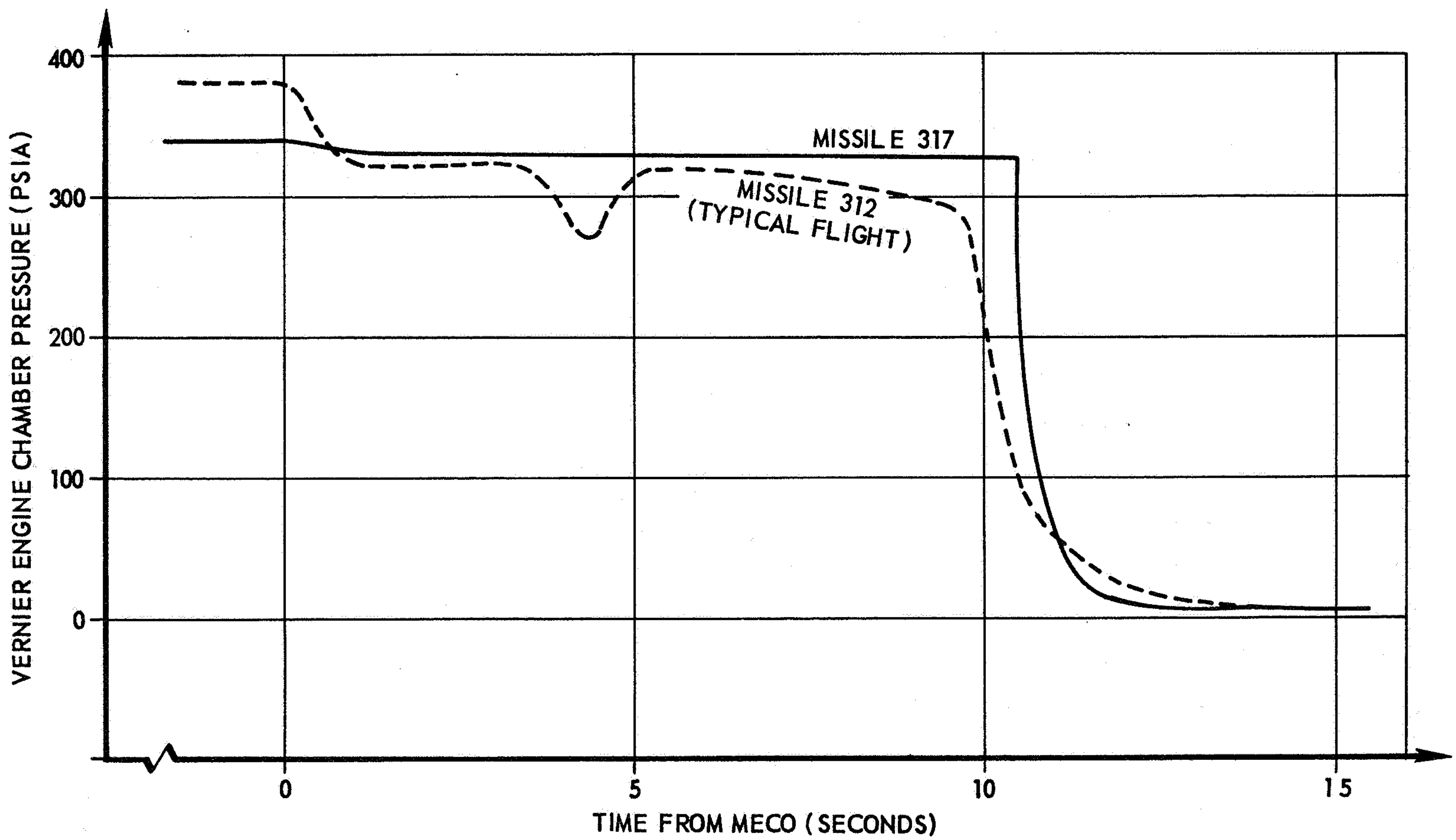

FIGURE 15

VERNIER ENGINE #1 CHAMBER PRESSURE TRANSDUCER SYSTEM

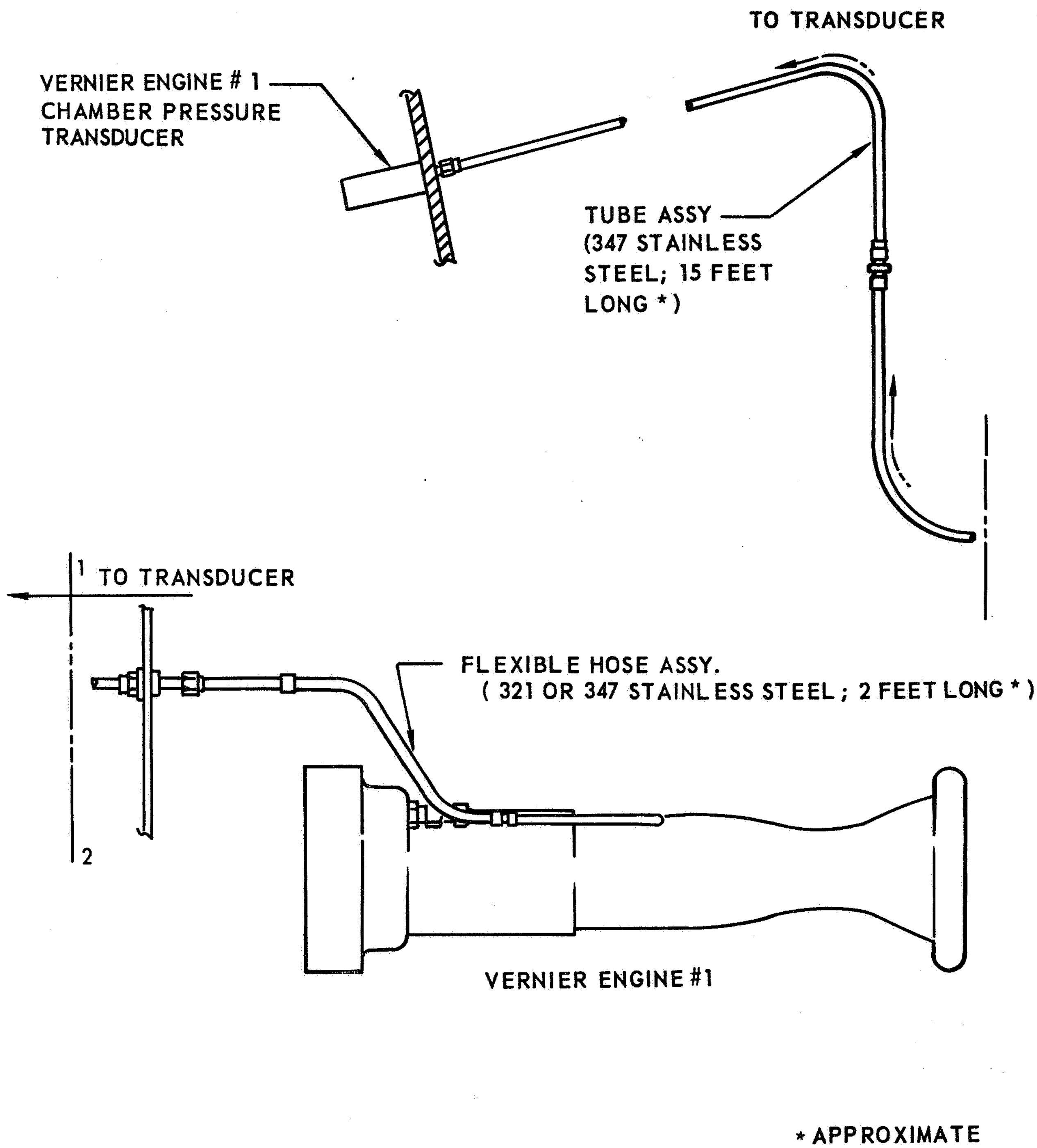

FIGURE 16

SECOND / THIRD STAGE SEPERATION

DISTANCE VS TIME FOR

TWO RETRO NOZZLES

THOR DELTA 317/2020/3020

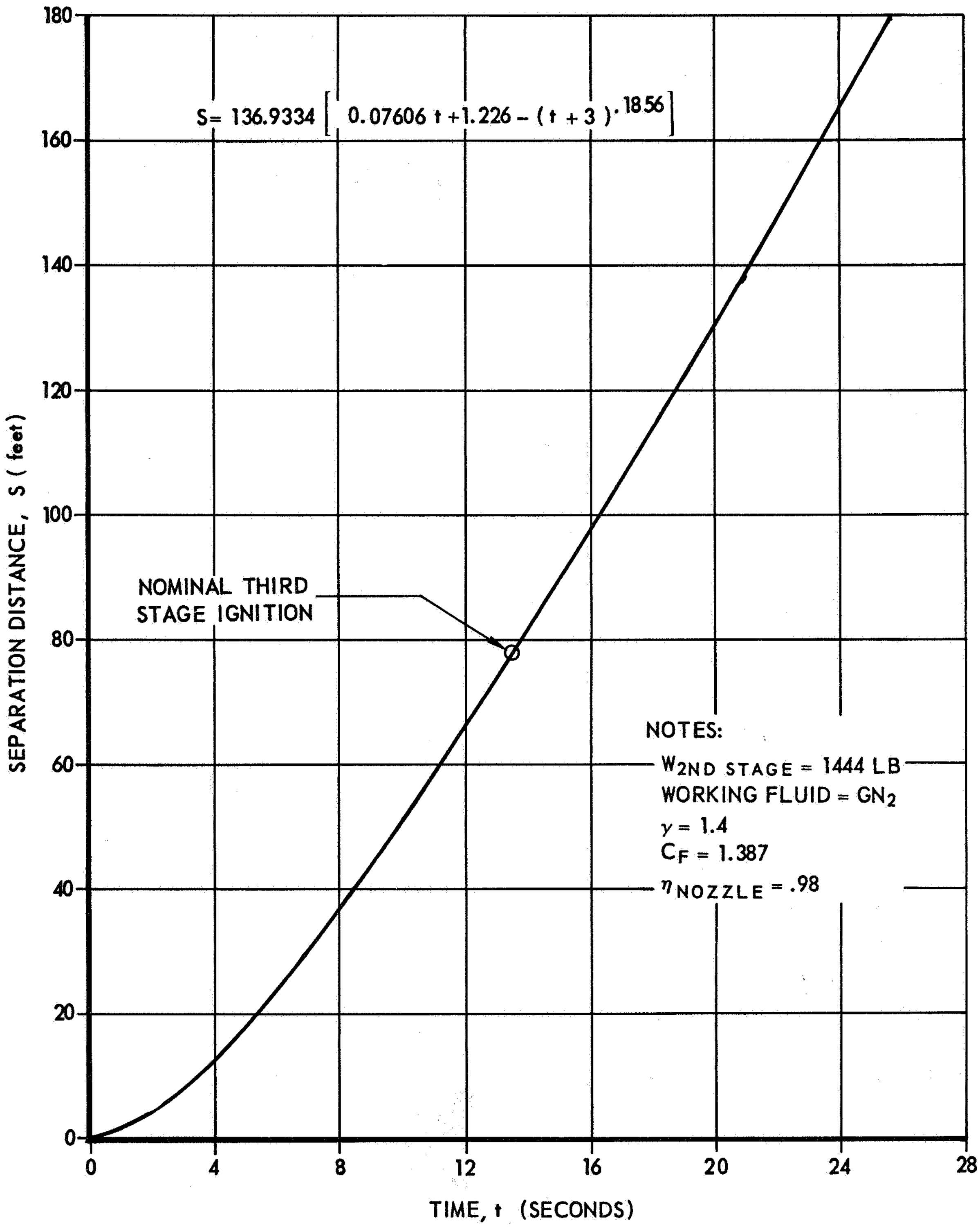

FIGURE 17

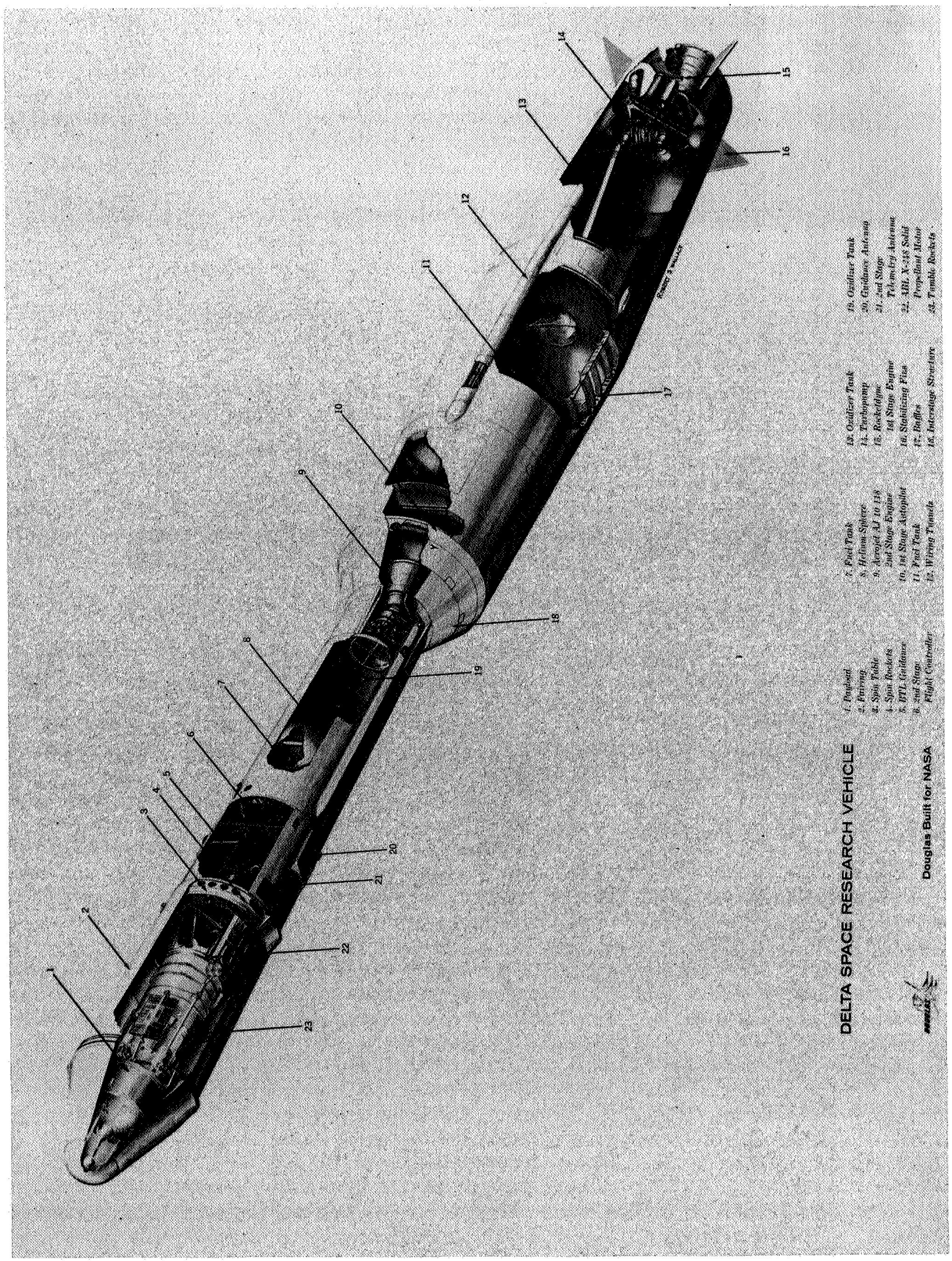
DELTA SPACE RESEARCH VEHICLE
1. Payload
2. Fairing
3. Spin Table
4. Spin Rockets
5. BTL Guidance
6. 2nd Stage Flight Controller
7. Fuel Tank
8. Helium Sphere
9. Aerojet AJ 10 118 2nd Stage Engine
10. 1st Stage Autopilot
11. Fuel Tank
12. Wiring Tunnels
13. Oxidizer Tank
14. Turbopump
15. Rocketdyne 1st Stage Engine
16. Stabilizing Fins
17. Baffles
18. Interstage Structure
19. Oxidizer Tank
20. Guidance Antenna
21. 2nd Stage Telemetry Antenna
22. ABL X-248 Solid Propellant Motor
23. Tumble Rockets
Douglas Built for NASA

FIGURE 18

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